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AFAPL-TR-72-17

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ADVANCED PROPULSION TECHNOLOGY ASSESSMENT FOR AN EXTERNALLY BLOWN FLAP TRANSPORT

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TECHNICAL REPORT AFAPL-TR-72-17

April 1972

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Air Force Aero Propulsion Laboratory
Air Force Systems Command
Wright-Patterson Air Force Base, Ohio

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FOREWORD

The work described in this report was conducted within the Propulsion Branch, AFAPL/TBA, of the Turbine Engine Division at the Air Force Aero Propulsion Laboratory, Wright-Patterson Air Force Base, Ohio. The effort was accomplished under Project 698DE from May 1970 through January 1971.

This report was submitted by the authors February, 1972.

Publication of this report does not constitute Air Force approval of the report's findings or conclusions. It is published only for the exchange and stimulation of ideas.

ERNEST C. SIMSPON
Director, Turbine Engine Division
Air Force Aero Propulsion Laboratory

Abstract

Results of this study indicate that, for an externally blown flap transport aircraft and missions investigated, aircraft gross weight reductions on the order of 10% can be obtained from the utilization of turbofan engines incorporating Advanced Technology Components when compared to near term propulsion technology. Engine thrust/weight ratio was clearly the most significant propulsion design parameter in terms of providing aircraft weight reductions. Other propulsion parameters such as cruise SFC, bypass ratio, and overall pressure ratio had only secondary effects on aircraft gross weight. While the effect of noise abatement was not considered, variations of engine thrust/weight ratio and cruise SFC were evaluated. Using these variations, preliminary estimates of the penalties associated with noise can be obtained by expressing it in terms of an engine thrust/weight reduction and cruise SFC increase and assessing the resultant aircraft weight increase. A recommendation is made to initiate a preliminary design activity whose objective would be to define suitable, high thrust/weight turbofan propulsion systems for the 1980+ time period.

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LIST OF ABBREVIATIONS

BPR	Engine bypass ratio
C_D	Aircraft drag coefficient
C_{D_C}	Drag correction factor
C_{L_1}	Aircraft take-off lift coefficient
C_{X_1}	Thrust minus drag force coefficient along the flight path at take-off
C_{LMAX}	Maximum lift coefficient at take-off
FPR	Engine fan pressure ratio
L_f	Normal accelaration, measured at the aircraft center of gravity, g units (Load Factor)
M_N	Aircraft flight Mach number
N. M.	Nautical miles
OPR	Engine overall pressure ratio
OWE	Aircraft operating empty weight
q	Dynamic pressure (lb/ft^2)
S	Aircraft wing reference area (ft^2)
SFC	Thrust specific fuel consumption
T	Net engine thrust
TIT	Engine turbine inlet temperature
TOGW	Aircraft take-off gross weight
T/W	Aircraft or engine thrust to weight ratio
W/S	Aircraft wing loading at take-off
α	Aircraft angle of attack
u	Runway friction factor

SECTION I

INTRODUCTION

In May 1970, the Tactical Air Command issued Required Operational Capability (ROC) No. 52-69, "Medium STOL Transport," which established a requirement for a new aircraft to replace the C-130. This aircraft is envisioned as a four engine, high subsonic speed, STOL transport capable of safe, routine operations onto 2000 foot runways. Projected initial operational capability (IOC) date for this aircraft varies from 1978 to 1983, with funding availability for full scale development being a critical parameter in establishing this variance.

Using mission, speed, and payload data from this ROC a parametric study was conducted by the Air Force Aero Propulsion Laboratory (AFAPL) to investigate the impact and influence of engine cycle parameters and propulsion state-of-the-art on the take-off gross weight characteristics of a typical STOL aircraft configuration. The principal objectives of the study were to evaluate the relative advantages of advanced versus near term propulsion technology for this class of aircraft and to define key propulsion items requiring advanced development demonstration effort. A turbofan powered, externally blown flap aircraft was selected for use in the study primarily because aerodynamic and weight data were available for this type of configuration. It is recognized that other aerodynamic lift concepts are being considered to satisfy the requirements for this mission which could result in the identification of propulsion configurations other than those discussed in this report.

SECTION II

MISSION DESCRIPTION

1. INTRODUCTION

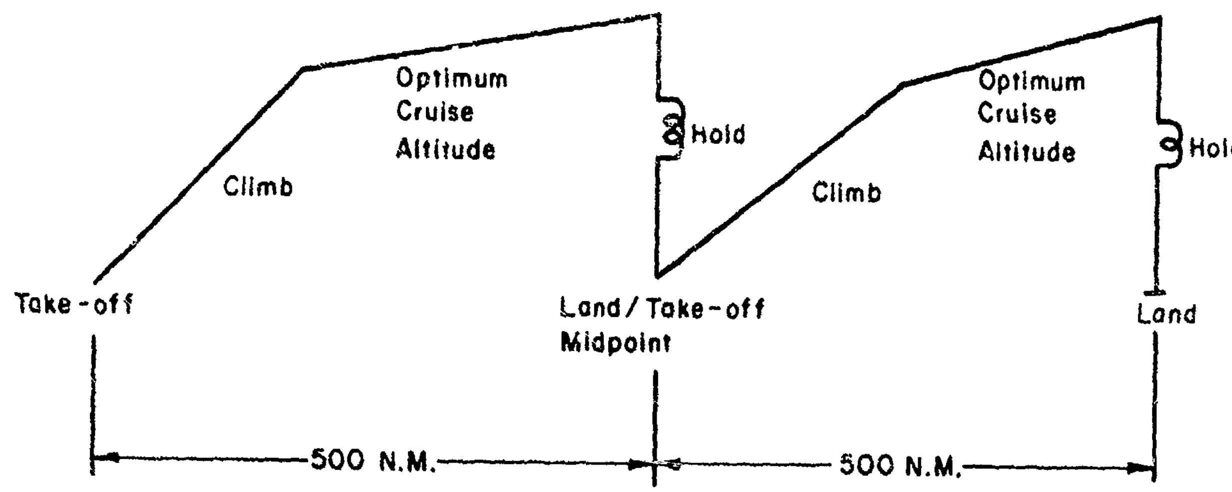
Two mission profiles, simulating a primary employment mission and a ferry deployment mission, were investigated in this study. Information required to formulate mission characteristics such as take-off field length, range, speed, payload, etc., was obtained from TAC ROC No. 52-69, "Medium STOL Transport," and from information obtained from the AFAPL Plans Office.

2. PRIMARY EMPLOYMENT MISSION

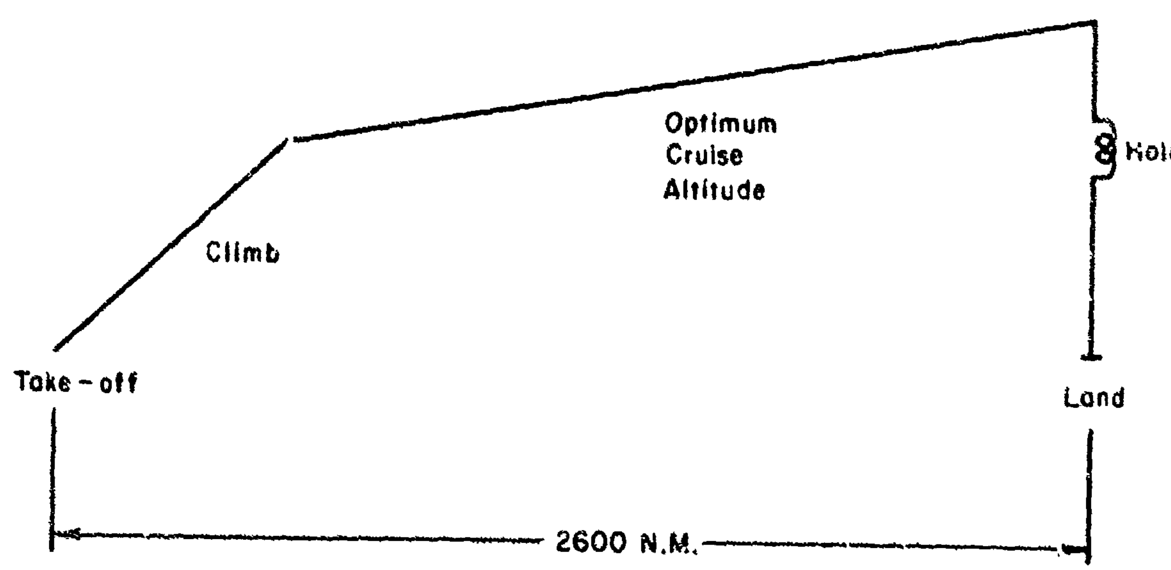
The primary mission profile is illustrated in Figure 1a. This mission was used to size the aircraft. Required payload is 28,000 pounds which is carried on both the outbound and return legs. Only internal wing fuel is utilized with no refueling permitted at the mid-point. Aircraft hold times at the mid-point and final destination are 10 and 30 minutes, respectively. Hold flight condition is $M_N = 0.2$ at 1000 feet altitude. All take-offs are according to the design requirements identified in Paragraph 4. Fuel consumed during start-up, warm-up, and take-off is based on all engines operating for one minute at maximum take-off power and five minutes at maximum continuous power.

3. DEPLOYMENT MISSION

The deployment or ferry mission is illustrated in Figure 1b. Desired payload is 38,000 lbs. This mission is flown after the aircraft has been sized for the employment mission and is used to determine the useful payload available. Aircraft hold time is 30 minutes and the load factor, L_f, is reduced from 3.0 as used in the primary mission, to 2.5.



a. Primary Mission



b. Ferry Mission

(Both Missions Flown At Standard Day Conditions)

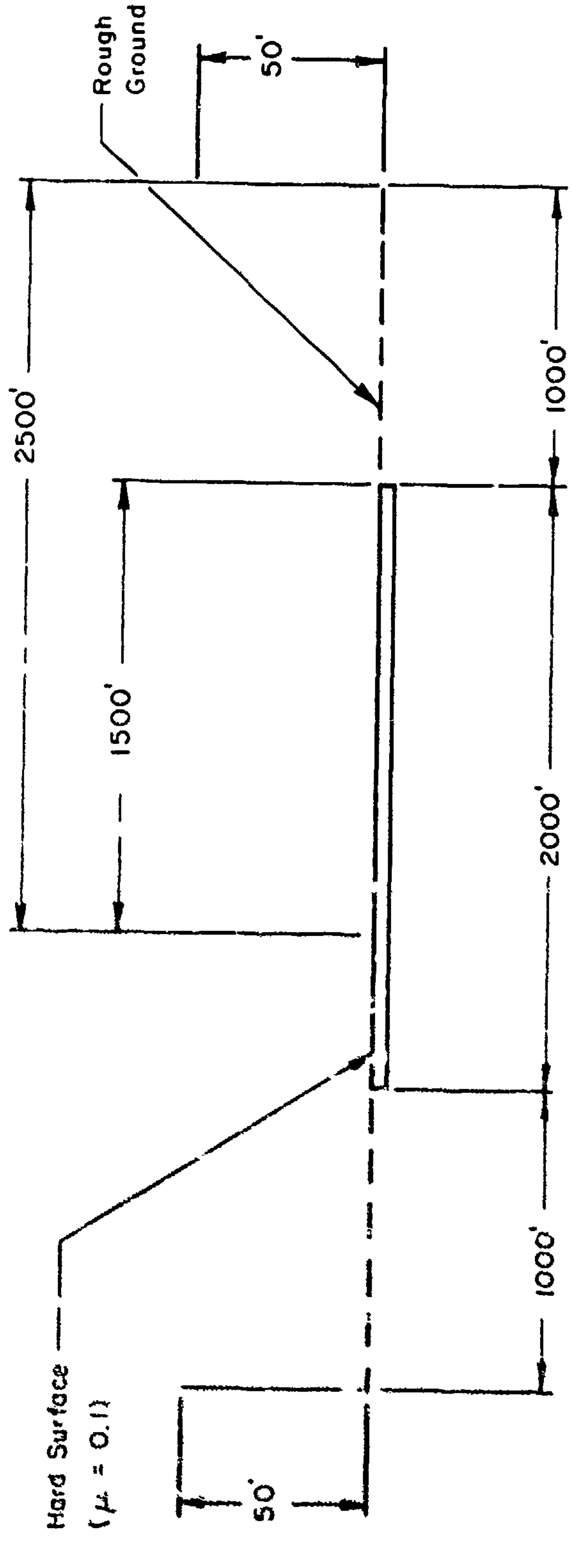
Figure 1. Medium STOL Transport Missions

4. TAKE-OFF REQUIREMENTS

Primary mission take-off requirements are illustrated in Figure 2. All four engines are used at maximum power during take-off. A runway length of 2000 ft. was utilized with the first 500 ft. arbitrarily assumed as being unavailable for use during take-off in order to provide a safety margin for uncertainties. The 50 ft. obstacle was placed 1000 ft. from the end of the runway. These assumptions result in a total take-off distance requirement of 2500 ft. to clear a 50 ft. obstacle, with a maximum of 1500 ft. being available for ground roll. All take-off conditions are at 2500 ft. altitude, 93.3°F day environment, and maximum aircraft gross weight. The propulsion systems are sized by the take-off requirement.

5. CRUISE SPEED REQUIREMENTS

ROC requirements state that the speed capability of this aircraft must be sufficient to insure theater arrival at or before that of strategic airlift in order to provide timely airlift support to air and ground forces. An altitude cruise speed of at least Mach 0.75 is specified. All cruise flight segments in the study were therefore flown at a speed of Mach 0.75.



Engines sized to meet 2500' take-off requirement

Runway located at 2500' altitude and hot day conditions take-off requirements

Figure 2. Runway Dimensions

SECTION III

AIRCRAFT CHARACTERISTICS

1. VEHICLE DESCRIPTION

The basic aircraft configuration selected for use in the study is shown in Figure 3. It is a moderately swept wing, tee-tail design, powered by four pod mounted turbofan engines. The engine nacelles are mounted in close proximity to the fuselage to minimize engine out control problems. High lift capability is achieved through the use of an externally blown flap arrangement.

2. AERODYNAMIC CHARACTERISTICS

a. Take-off

The use of externally blown flaps results in the take-off lift coefficient becoming a function of engine gross thrust coefficient as well as angle of attack. Figure 4 shows the aerodynamic take-off characteristics as functions of installed gross thrust coefficient and angle of attack. All lift coefficients used in the mission program were reduced according to the following relationships to account for the preliminary nature of the data used and to provide margin for potential aircraft stability and control problems.

$$C_{L_{curve}} = C_L \text{ Obtained from Figure 4}$$

$$C_{L_{corr.}} = C_L \text{ Used in Mission Program}$$

FOR $0 < C_{L_{curve}} < 4.5$

$$C_{L_{corr.}} = 0.95 \times C_{L_{curve}}$$

FOR $4.5 < C_{L_{curve}} < C_{L_{max}}$

$$C_{L_{corr.}} = .95 - \frac{.05\,(C_{L_{curve}} - 4.5)}{C_{L_{max}} - 4.5}\; C_{L_{curve}}$$

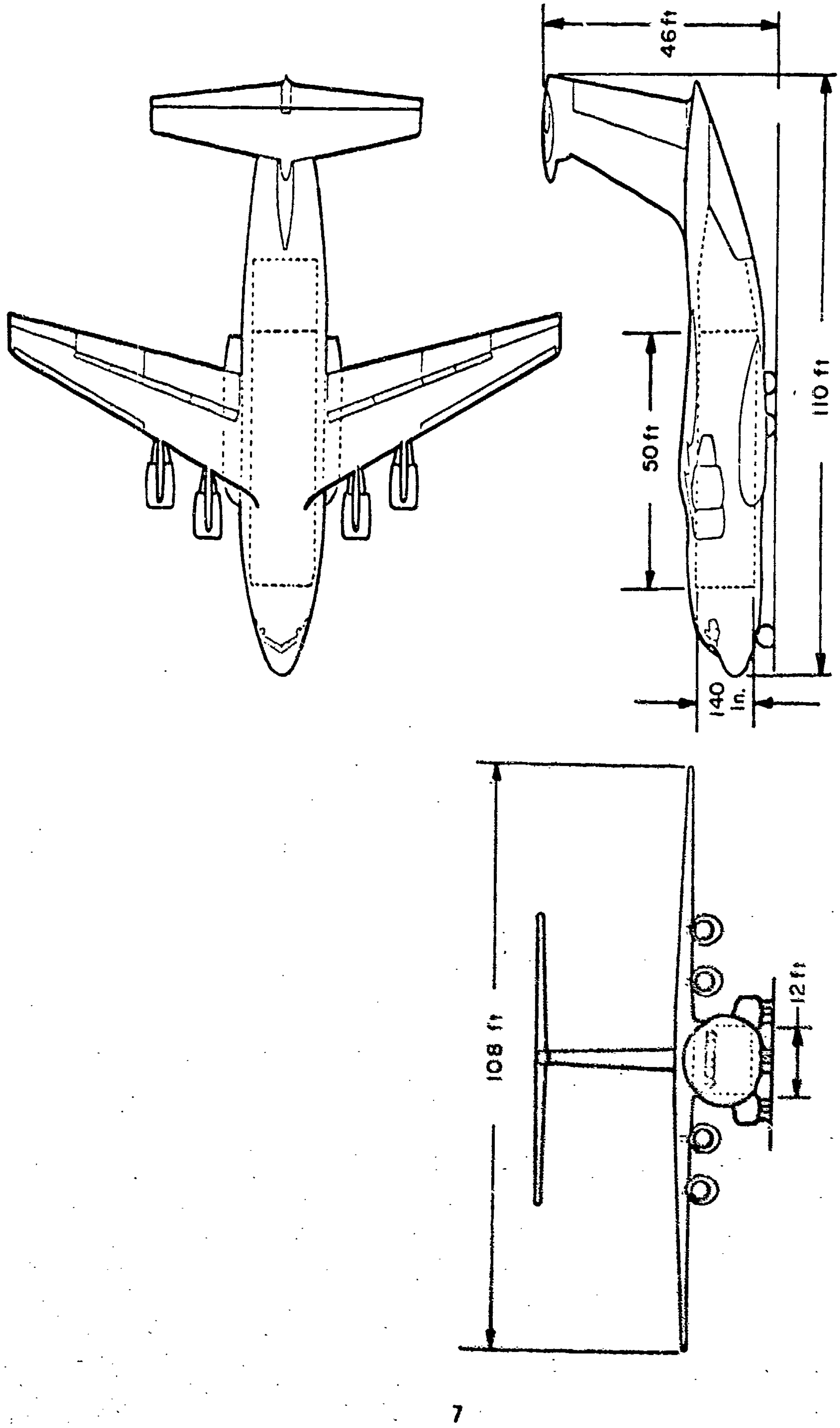


Figure 3. Basic Aircraft Configuration

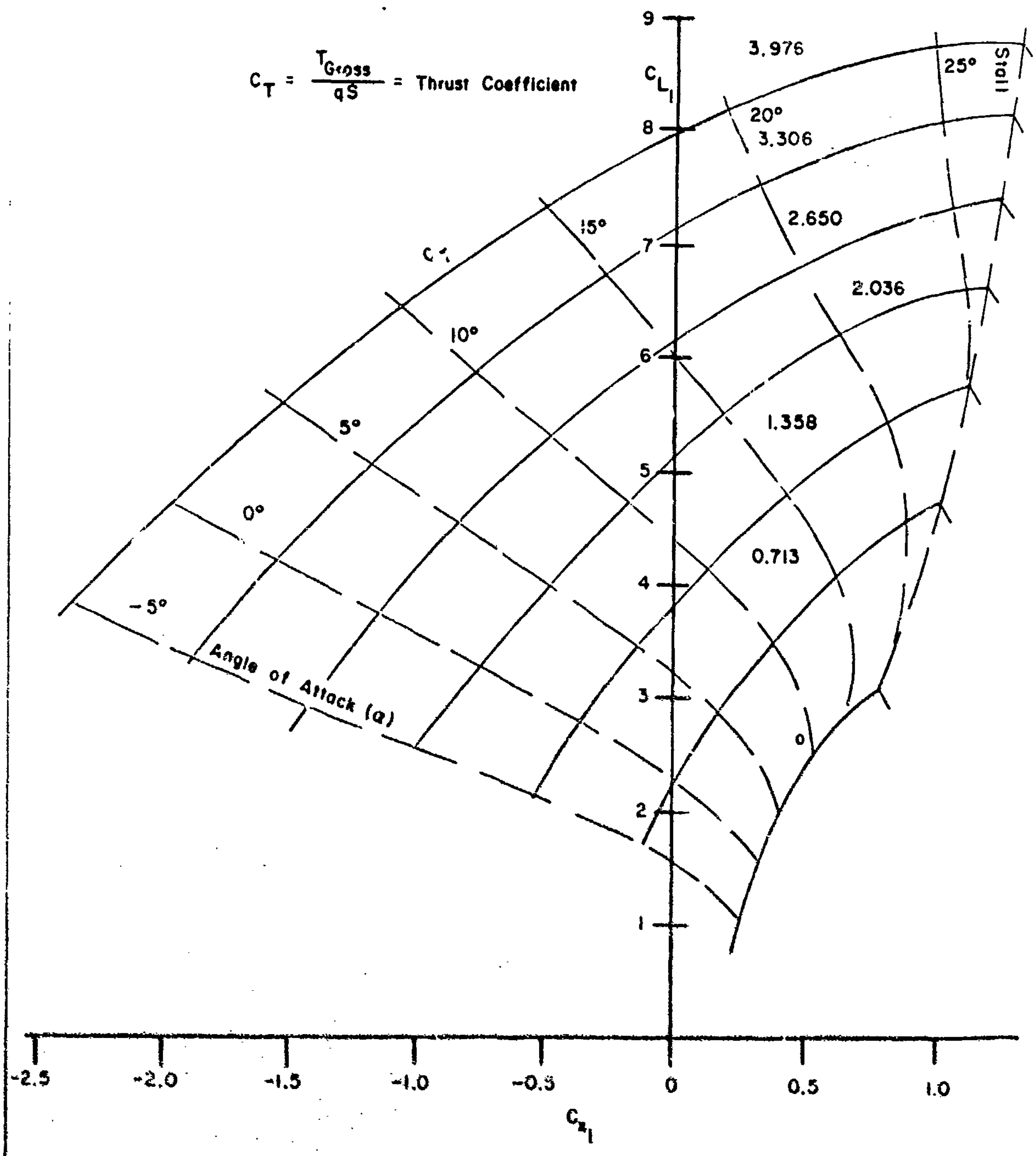


Figure 4. Aerodynamic Data for an Externally Blown Flap Aircraft

A complete discussion of the take-off program is given in Reference 1.

b. Cruise

Cruise performance assessment was accomplished by means of the mission analysis program described in Reference 2. The assumed lift and drag characteristics as a function of Mach number and angle of attack are illustrated in Figures 5 and 6. These characteristics were assumed constant and were not corrected for Reynold's number effects as the aircraft wing loading was varied.

3. AIRCRAFT WEIGHT AND SCALING

Figure 7 shows the variation of basic aircraft operating weight empty (OWE), installed propulsion system weight, and wing weight as functions of aircraft take-off wing loading. As indicated, significant changes in take-off wing loading result in only a small variation in aircraft operating weight empty. Selection of wing loading will result in establishing an operating weight empty for the baseline aircraft. This process is described in Paragraph 4.

As the various parametric engines are installed on the basic aircraft, the aircraft must be resized to account for differences in individual engine performance (fuel consumption), size, and weight. Discussions with aircraft manufacturers resulted in the definition of a Δ take-off gross weight factor of 1.87 to account for these differences. The application of this factor results in a total take-off gross weight reduction of 1.87 pounds for every pound of weight removed from the propulsion system and/or fuel load. This factor remains constant for the range of take-off gross weights being considered in this study. Appendix I presents a general outline of the aircraft and engine sizing technique.

4. AIRCRAFT WING LOADING SELECTION

Having established the aircraft configuration and aerodynamics; plus the requirements imposed by the missions, only the aircraft wing loading and thrust loading remain unspecified.

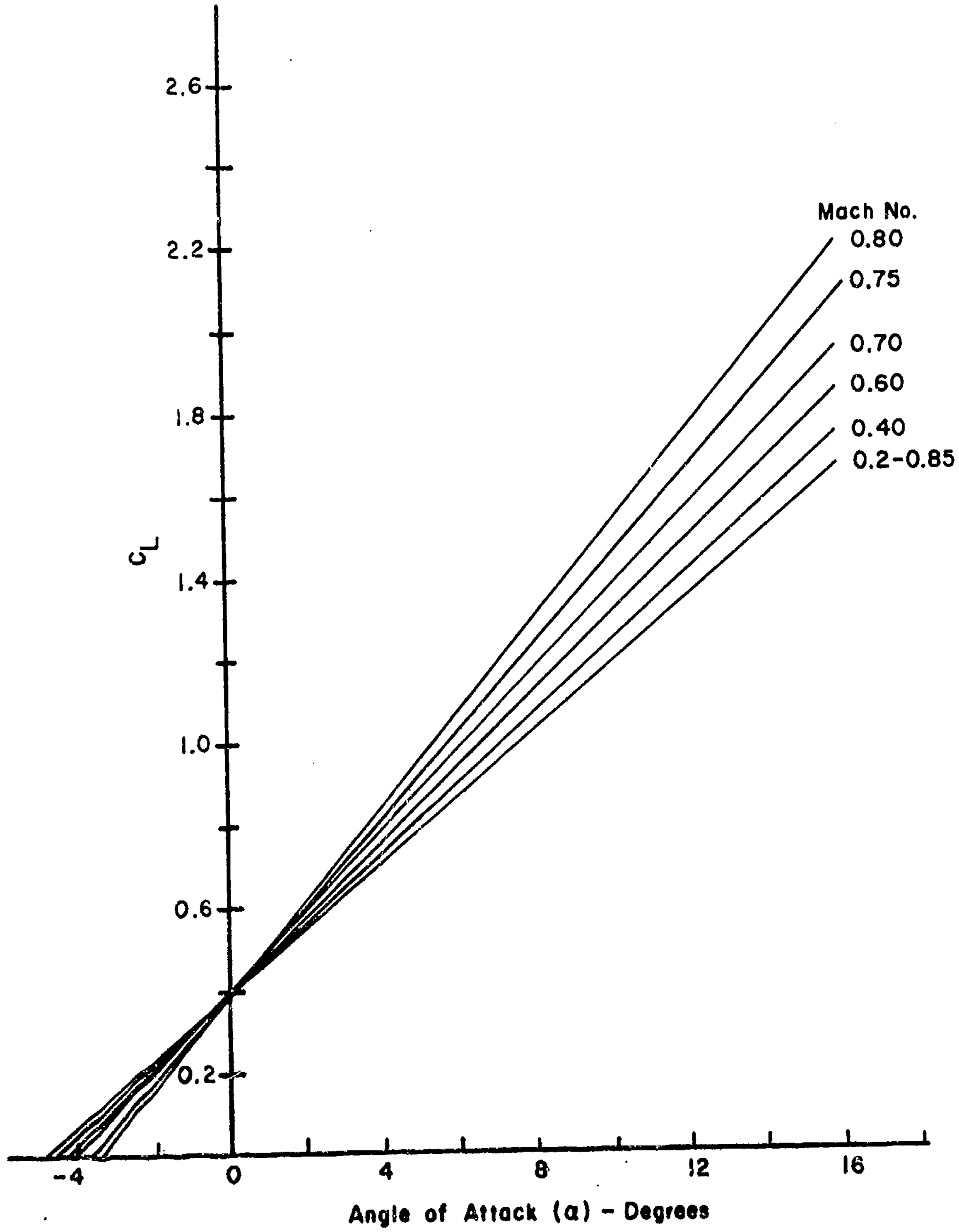


Figure 5. Lift Characteristics

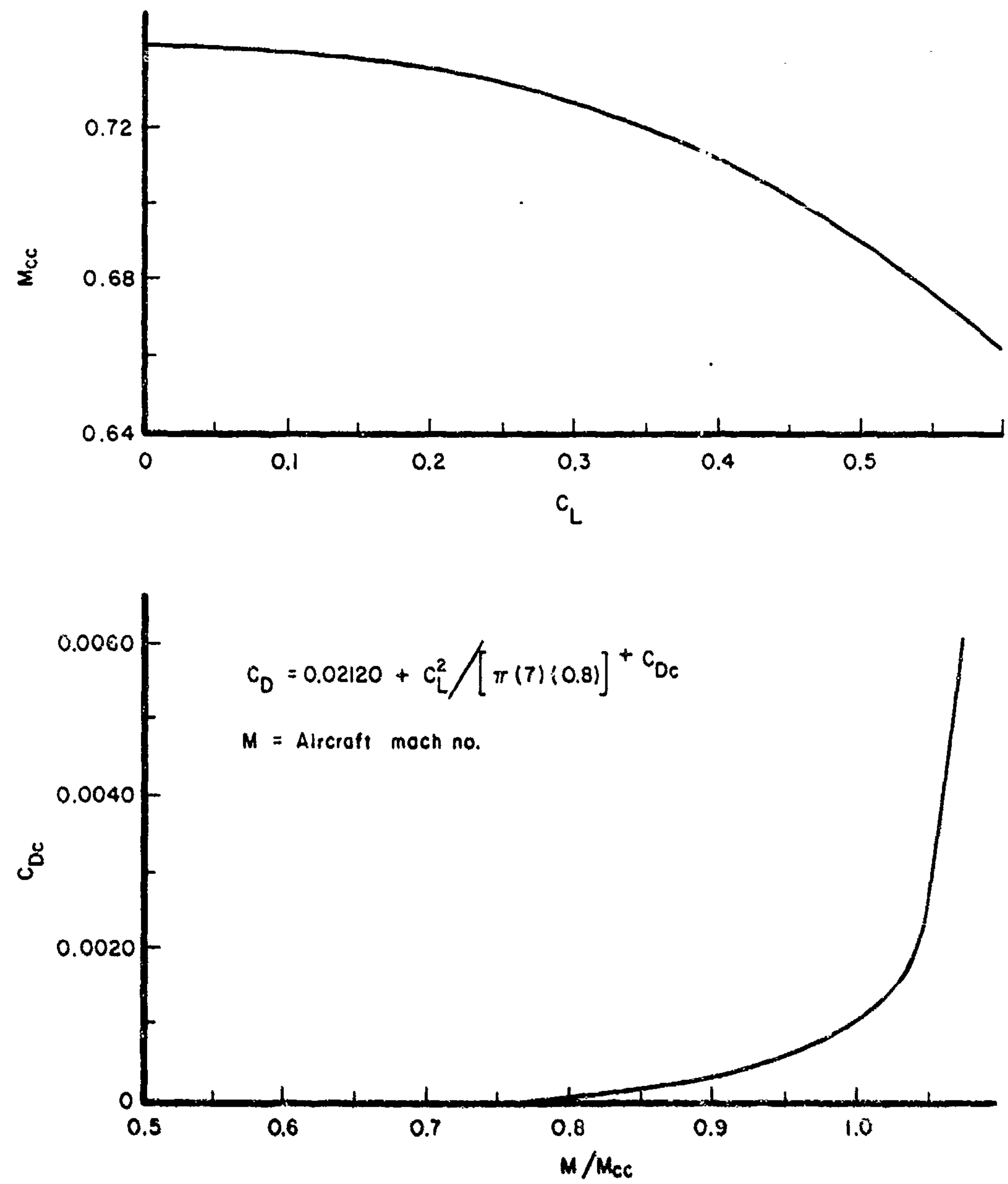


Figure 6. Drag Characteristics

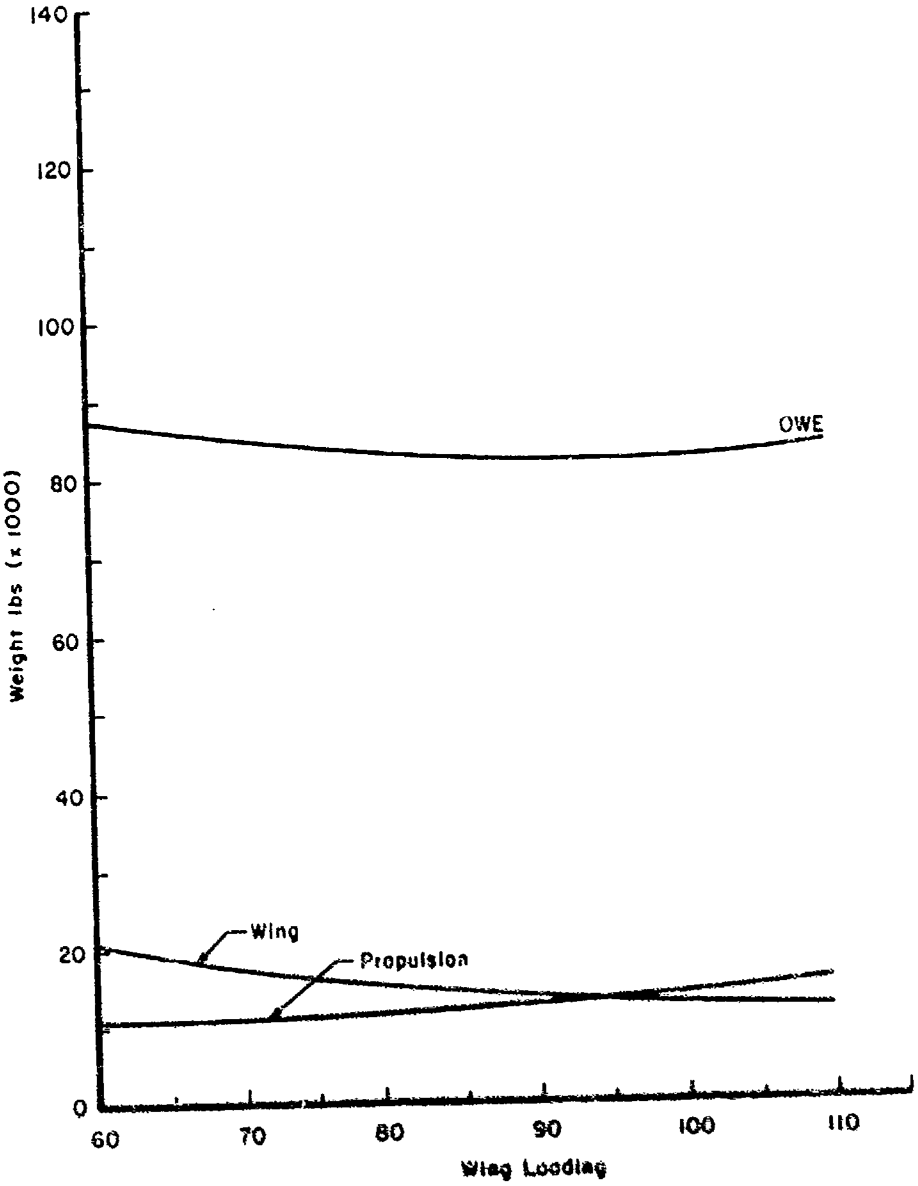


Figure 7. Base Point Aircraft and Propulsion Sizing Characteristics

A preliminary analysis was conducted to determine a representative value for aircraft wing loading. This value was then fixed for all parametric engine evaluations. Since this study is concerned only with the relative effect of advanced engine technology on aircraft take-off gross weight, the selection of a fixed wing loading should not impose any significant compromises on the relative effect of advanced technology engines. However, it is probable that complete optimization of the aircraft and propulsion system characteristics would result in greater relative advantages for aircraft using advanced technology engines.

The preliminary analysis was carried out using contractor study engine bulletins. The characteristics of each engine are listed in Table 1. Plots of aircraft thrust/weight (T/W) versus aircraft wing loading and take-off distance were generated using the take-off computer program described in Reference 1. Figures 8 and 9 depict the results of the analysis for a contractor test engine No. 1. Several primary mission runs were made at various wing loadings for each test engine. The mission analysis program used to make these runs is described in Reference 2. The results of this work are shown in Figure 10.

A take-off wing loading of 97.5 lbs/ft^2 was selected as the best compromise between overall mission performance and aircraft stability and control requirements at take-off.

TABLE I

TEST ENGINES

TEST ENGINES	No. 1	No. 2	No. 3
Bypass Ratio	7.6	6.55	6.0
Overall Pressure Ratio	17.0:1	23.07	19.0:1
Turbine Inlet Temp. °F	2280	2450	2400
Basic Engine Thrust/Weight	7.40	7.61	7.44
SFC - SLS-STD DAY	0.335	0.352	0.372

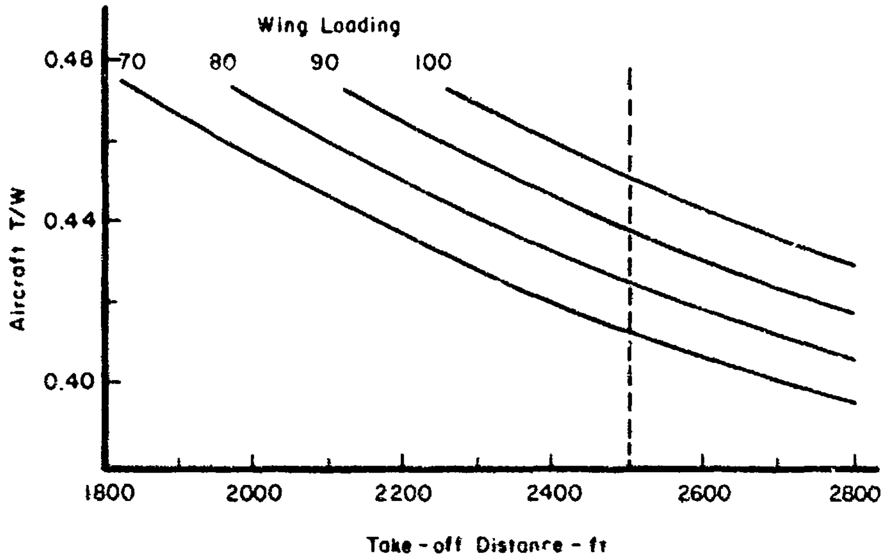


Figure 8. Aircraft Thrust/Weight as a Function of a Wing Loading and Take-Off Distance

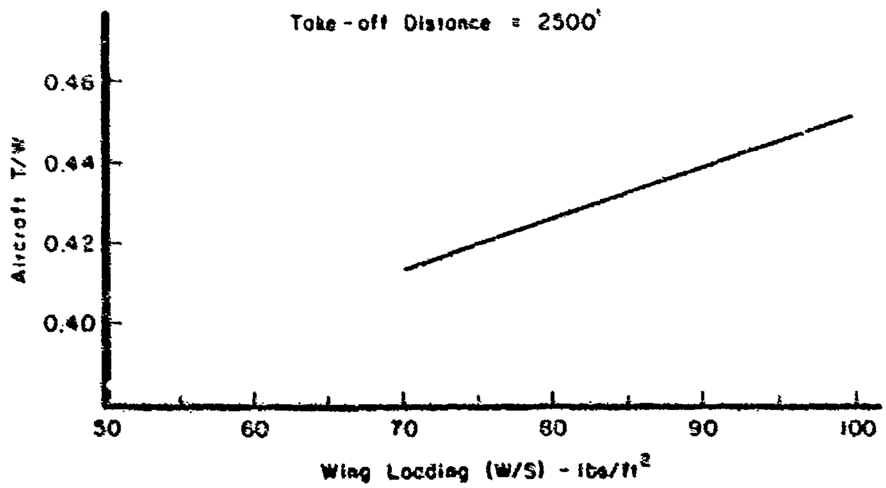


Figure 9. Required Aircraft Thrust/Weight VS. Wing Loading

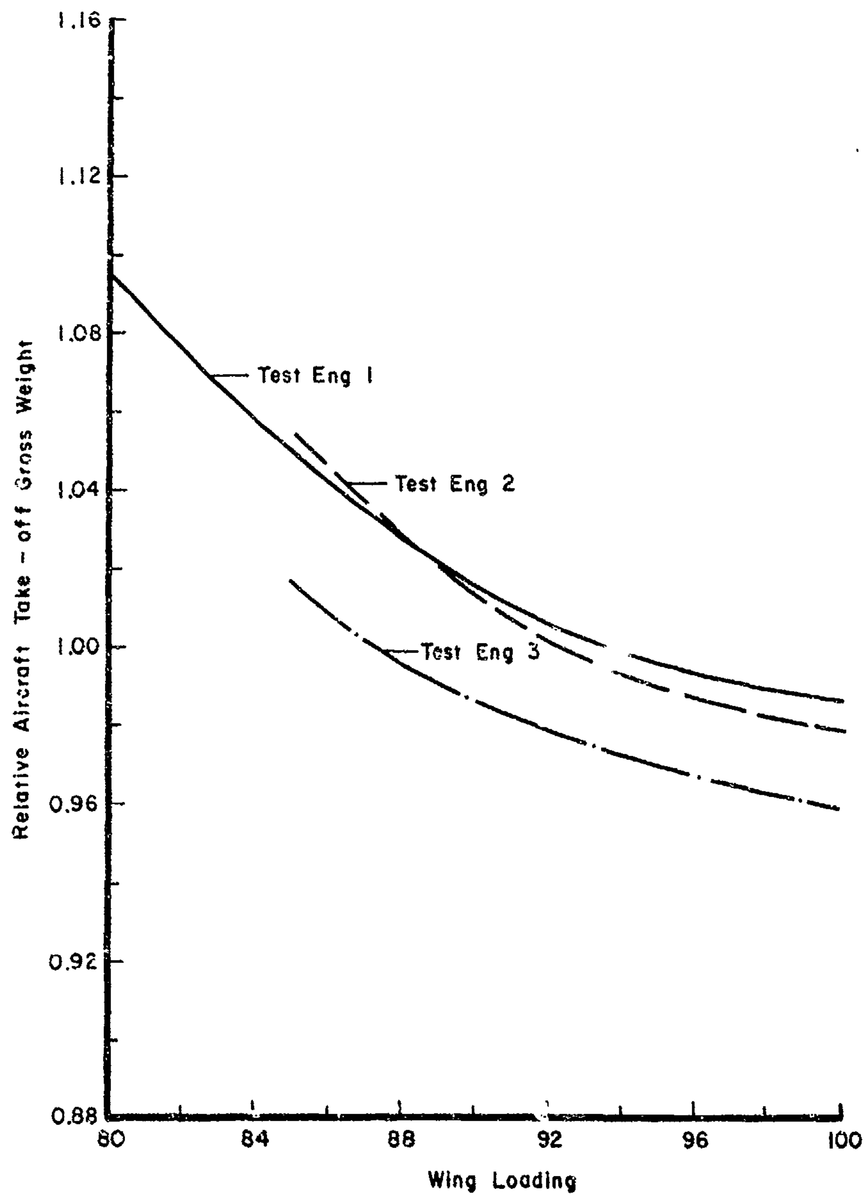


Figure 10. Variation of Take-Off Gross Weight With Wing Loading

SECTION IV

PROPULSION

1. INTRODUCTION

The engines evaluated in the parametric study were dual spool turbofans having fixed primary and fan duct nozzles, see Table II. All engines were initially sized for a sea level, hot day airflow of 390 lbs/sec, and were flat rated to sea level, 90°F conditions. Noise constraints were not considered during the study since the Air Force has not established requirements in this area. When firm noise criteria or requirements have been identified a reassessment should be undertaken to determine the effect on engine cycle selection and overall aircraft performance and weight. Two propulsion technology state-of-the-art levels were considered in the study and the range of cycle parameters investigated for each technology level are listed below:

	Baseline Technology	Advanced Technology
Max. Turbine Rotor Inlet Temperature (°F)	2450°	2650°, 2850°, 3050°
Bypass Ratio (BPR)	3.5 - 7.5	4.5 - 11.5
Overall Pressure Ratio (OPR)	20 - 28	23 - 34

2. PROPULSION TECHNOLOGY DEFINITION

a. Baseline Technology

This technology level is representative of turbofan engines that could complete model qualification tests (MQT) in the 1974-1975 time period. A maximum turbine inlet temperature (TIT) of 2450°F was selected as being appropriate for this level of technology.

TABLE II

PARAMETRIC ENGINES

BASELINE TECHNOLOGY		ADVANCED TECHNOLOGY					
TIT = 2450°F		TIT = 2650°F		TIT = 2850°F		TIT = 3050°F	
OPR = 20:1 FAN PRESSURE RATIO		OPR = 23:1		OPR - 23:1		OPR - 23:1	
BPR	(FPR)	BPR	FPR	BPR	FPR	BPR	FPR
3.5	1.94	4.5	1.95	4.5	2.2	7.0	1.83
4.5	1.72	5.5	1.80	5.5	1.88	8.5	1.66
6.5	1.55	6.5	1.66	7.0	1.66	10.0	1.56
		7.5	1.57	8.5	1.57	11.5	1.46
		8.5	1.5	10.0	1.50		
OPR = 23:1		OPR = 28:1		OPR = 28:1		OPR = 28:1	
BPR	FPR	BPR	FPR	BPR	FPR	BPR	FPR
3.5	1.94	4.5	1.95	4.5	2.2	7.0	1.80
4.5	1.72	5.5	1.80	5.5	1.88	8.5	1.66
5.5	1.64	6.5	1.66	7.0	1.66	10.0	1.56
6.5	1.54	7.5	1.57	8.5	1.57	11.5	1.46
7.5	1.44	8.5	1.48	10.0	1.50		
OPR = 28:1		OPR = 34:1		OPR = 34.1		OPR = 34:1	
BPR	FPR	BPR	FPR	BPR	FPR	BPR	FPR
6.5	1.5	4.5	1.95	4.5	2.0	7.0	1.79
		5.5	1.70	5.5	1.8	8.5	1.65
		6.5	1.60	7.0	1.65	10.0	1.53
		7.5	1.55	8.5	1.55	11.5	1.46
		8.5	1.45	10.0	1.45		

This technology level is currently being investigated by airframe and engine contractors to determine representative engine/airframe designs. For this reason a complete study of this technology level was not conducted. Rather, a few representative baseline engine designs were developed and a base point design selected which is in general agreement with current contractor work. This approach was implemented to insure that baseline and advanced technology engines could be evolved and evaluated on a consistent and comparable basis.

b. Advanced Technology

This technology level was established to represent turbofan engines that would initiate engineering development in the late 1975 to 1976 time period with completion of Model Qualification Tests (MQT) in the 1979-1980 time period. Composite materials, improved high-strength/temperature alloys, and higher stage loading rotating components would be utilized in these engines. Based on projected trends, maximum turbine inlet temperatures between 2650°F and 3050°F were defined as being feasible for this level of technology at varying degrees of risk.

3. CYCLE COMPONENT EFFICIENCIES AND LOSSES

The range of component efficiencies and pressure losses used to compute engine performance for both the baseline and advanced technology engine is listed in Table III. Identical aerodynamic component performance was used for both levels of technology based on the assumption that, for the mission being studied, advanced technology should be utilized to reduce overall engine weight and dimensions rather than attempting to significantly improve engine performance. This is expected to result in the use of higher tip speed/higher loaded rotating components which have efficiency levels that are comparable to current technology components.

TABLE III

DESIGN POINT COMPONENT EFFICIENCY AND PRESSURE LOSSES FOR BASELINE AND ADVANCED TECHNOLOGY ENGINES

Fan Efficiency Range		0.851 to 0.860
Compressor Efficiency Range		0.851 to 0.858
Combustor Efficiency		0.985
High Pressure Turbine Efficiency Range		0.875 to 0.878
Low Pressure Turbine Efficiency Range		0.915 to 0.920
Combustor Pressure Drop	2450°F Engines	0.048
	2650°F Engines	0.053
	2850°F Engines	0.058
	3050°F Engines	0.073
Fan-Compressor $\Delta P/P$		0.010
Fan Duct $\Delta P/P$		0.015
Gas Generator Duct $\Delta P/P$		0.005
Fan Nozzle Velocity Coefficient		0.997
Gas Generator Nozzle Velocity Coefficient		0.997

4. TURBINE COOLING AIRFLOW

For the baseline technology engines, chargeable turbine cooling airflow was set at 9% of compressor discharge airflow. This value appears to be consistent with proposed industry state-of-the-art engine configurations operating with similar turbine inlet temperatures. For the advanced technology engines, it was assumed that the use of improved high temperature alloys and advanced cooling techniques would allow turbine inlet temperatures to be increased by 200°F to 2650°F without increasing the required chargeable turbine cooling airflow. As turbine inlet temperature is increased above the 2650°F value, required turbine cooling airflow increases linearly at the rate of 1% for each 100°F increase. This rate represents an average of various contractor predictions in this area.

5. ENGINE PERFORMANCE

Uninstalled and installed performance for each engine cycle was generated using the SMOTE off-design point matching program (Reference 3) and a subsonic turbofan, nacelle installation program (Reference 4). Component maps, based on contractor test data, were utilized to compute performance for all engines. Accessory horsepower extraction and customer bleed effects were not considered, since requirements for these items are not available and it is estimated they would have only a small relative effect on the engine technology comparison.

A preliminary investigation was conducted to define optimum fan pressure ratios, based on minimum cruise SFC, for each of the advanced technology engine configurations. Results of the investigations are shown in Figures 11 through 13.

Uninstalled performance data for each engine was then generated at the take-off, climb, cruise and hold conditions. The thrust, SFC, and airflow output was then adjusted for installation effects based on a short duct nacelle arrangement and using the procedures described in Appendix II.

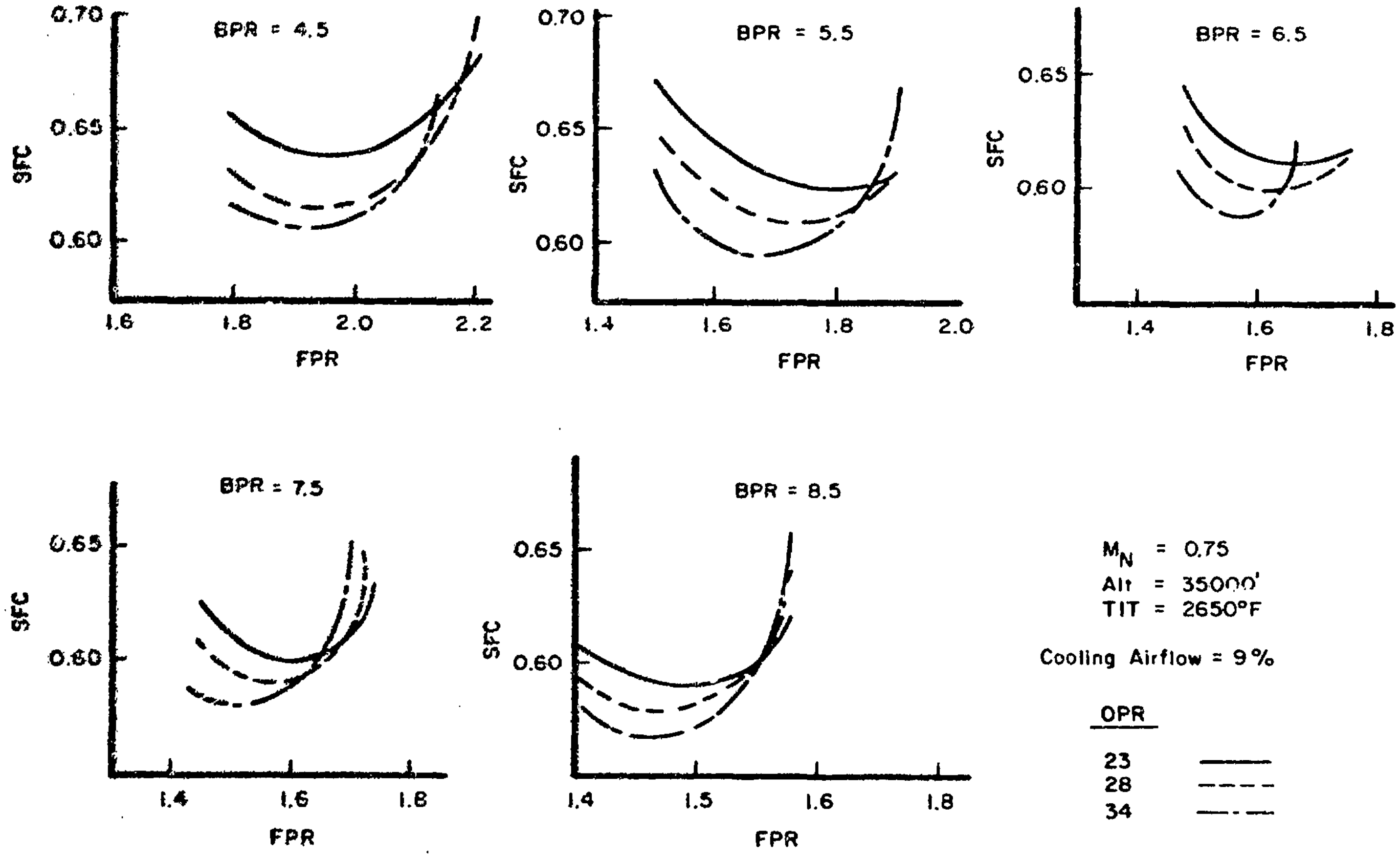


Figure 11. Crusie SFC Vs. Fan Pressure Ratio at 2650°F Turbine Inlet Temperature

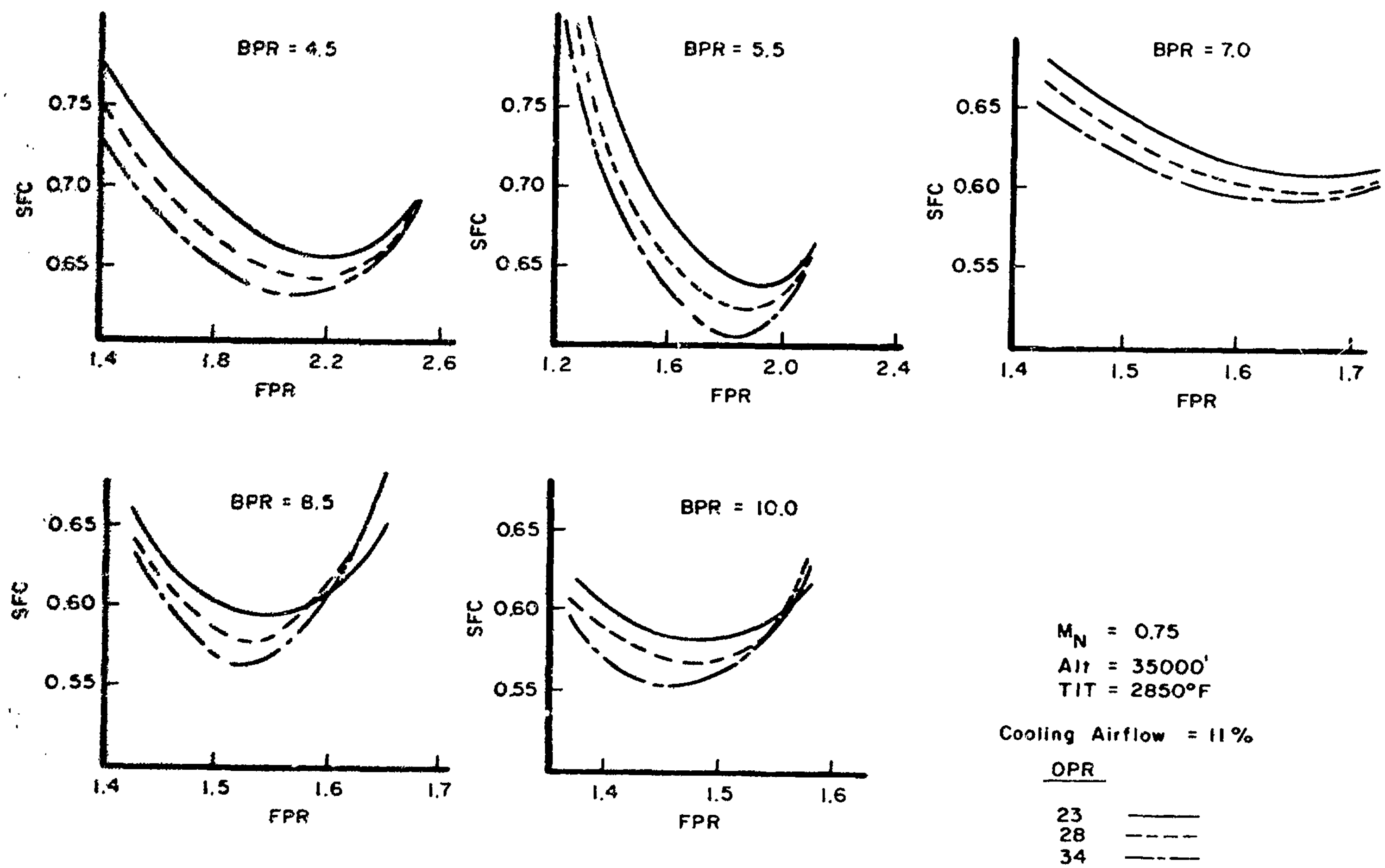


Figure 12. Cruise SFC Vs. Fan Pressure Ratio at 2850°F Turbine Inlet Temperature

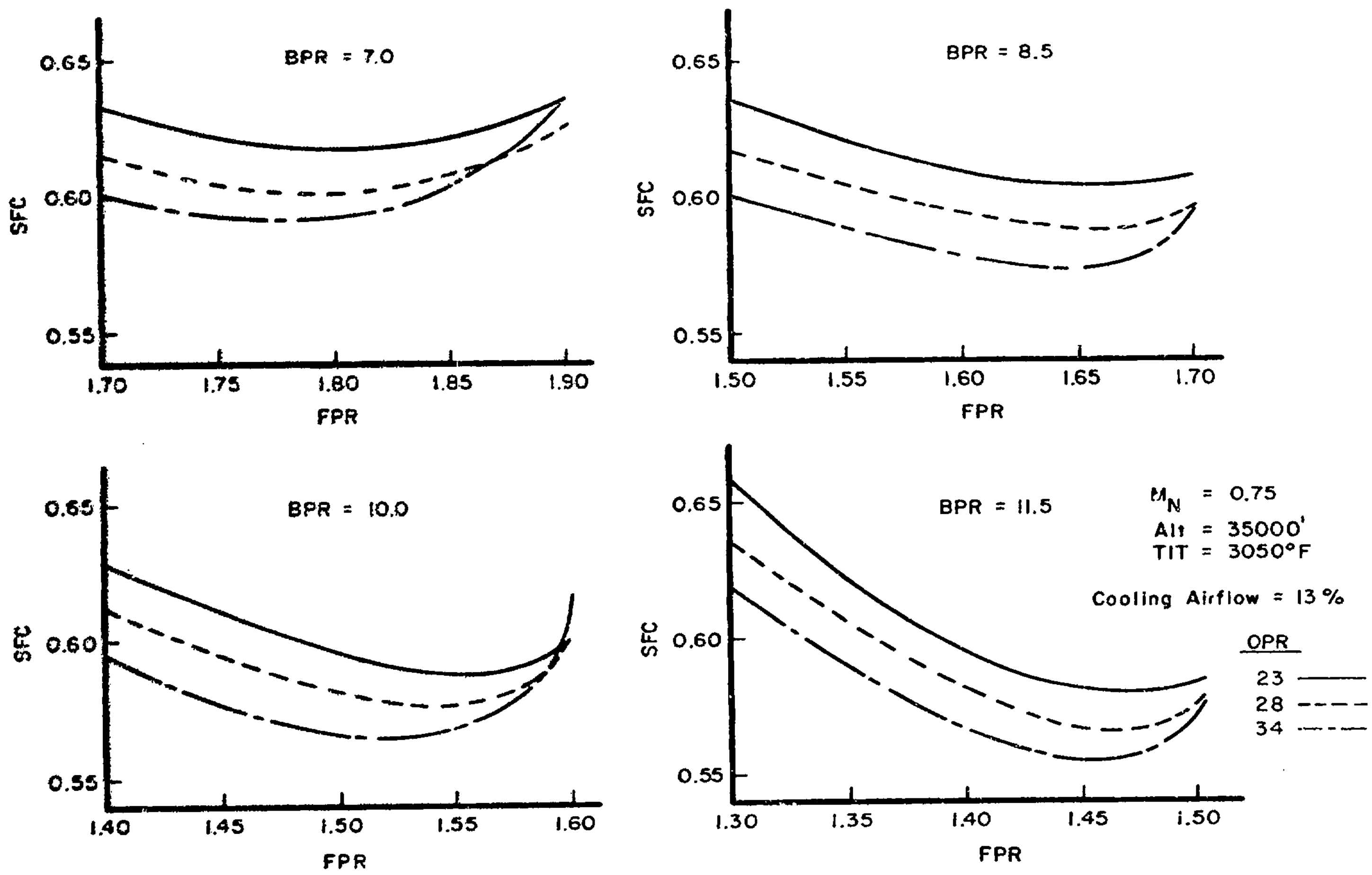


Figure 13. Cruise SFC Vs. Fan Pressure Ratio at 3050°F Turbine Inlet Temperature

6. PROPULSION SYSTEM WEIGHT AND SCALING

Accurate methods of estimating the uninstalled weight of advanced technology engines are not well developed, except through the use of layout or design drawings. The latter technique is normally not suitable for use in a parametric design study.

To overcome this situation three base engine thrust/weight levels were selected for each turbine inlet temperature, and the effect of engine thrust/weight variation was assessed for both the primary employment and ferry deployment missions. The selected base engine thrust/weight levels for each turbine inlet temperature are listed below:

Turbine Inlet Temperature	Base Engine Thrust/Weight
2450°F	5.5, 6.5, 7.5
2650°F	7.0, 8.0, 9.0
2850°F	8.0, 9.0, 10.0
3050°F	9.0, 10.0, 11.0

The base thrust/weight levels were adjusted for the effects of bypass ratio and overall pressure ratio, at a turbine inlet temperature, by means of the curves depicted in Figure 14 through 17. The trends established in these figures were derived from Reference 5.

After the base engine thrust/weight has been adjusted, each engine was scaled, as required, according to the following relationship:

$$\left.\frac{T}{W}\right)_{SCALED} = \left.\frac{T}{W}\right)_{REF} \times \left(\frac{T_{REF}}{T_{SCALED}}\right)^{0.2}$$

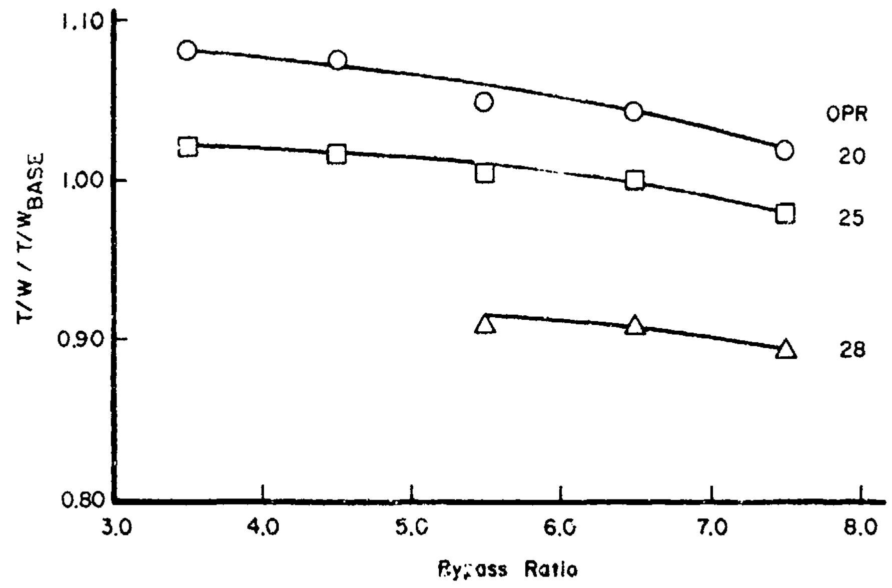


Figure 14. Variation of Thrust/Weight Ratio with Bypass Ratio and Overall Pressure Ratio at 2450°F Turbine Inlet Temperature

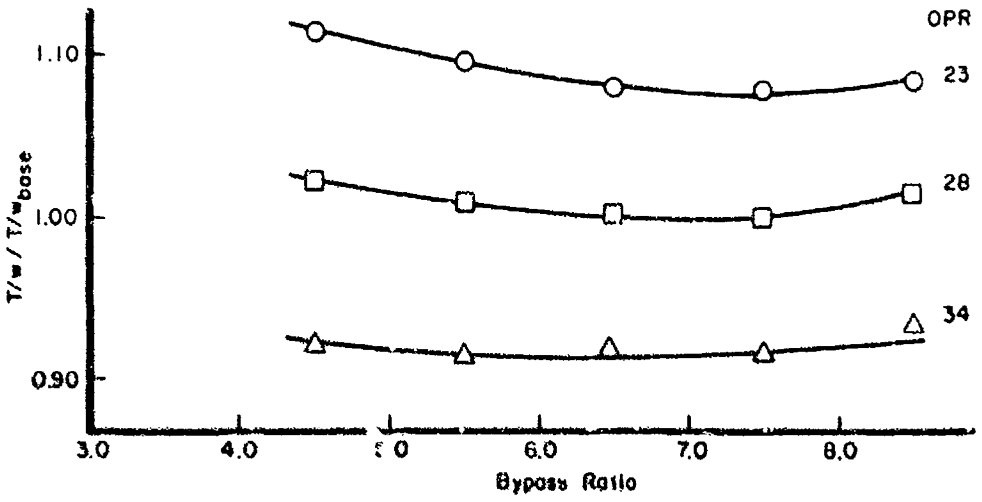


Figure 15. Variation of Thrust/Weight Ratio with Bypass Ratio and Overall Pressure Ratio at 2650°F Turbine Inlet Temperature

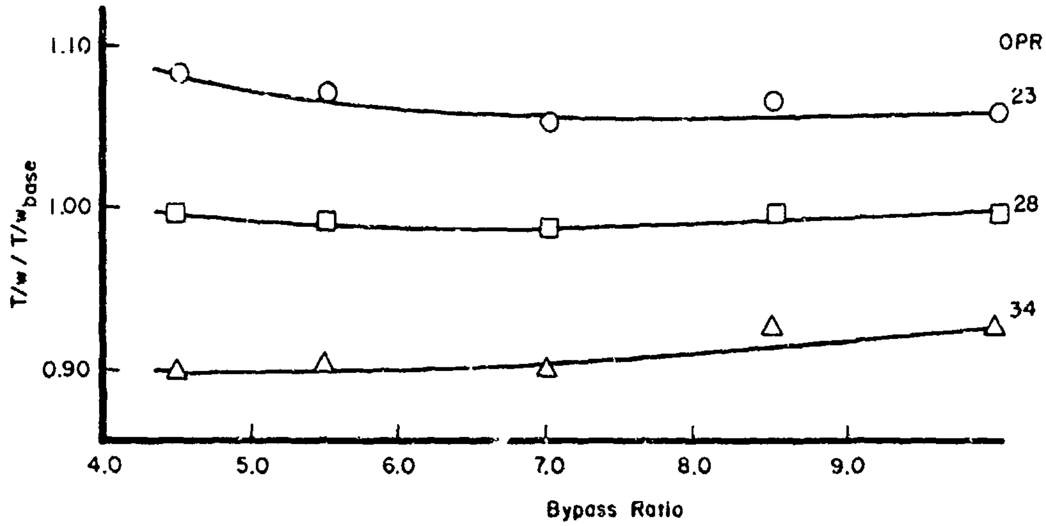


Figure 16. Variation of Thrust/Weight Ratio with Bypass Ratio and Overall Pressure Ratio at 2850°F Turbine Inlet Temperature

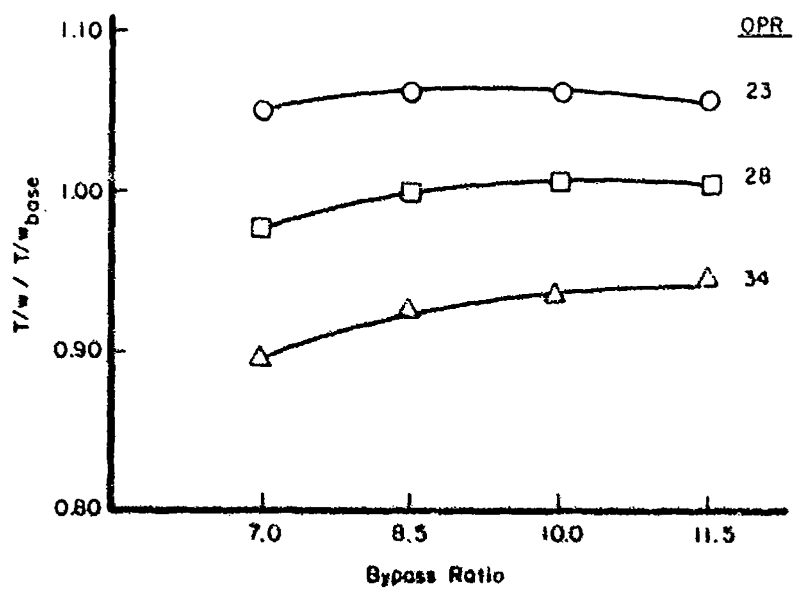


Figure 17. Variation of Thrust/Weight Ratio with Bypass Ratio and Overall Pressure Ratio at 3050°F Turbine Inlet Temperature

Where:

T_{REF} - Based on a sea level, hot day airflow at 390 lbs/sec.

W_I - Installed propulsion system weight.

T_{SCALED} - Installed thrust required by aircraft.

NOTE: The above relationship is based on the following equation:

$$\frac{W_{I_{REF}}}{W_{I_{SCALED}}} = \left(\frac{T_{REF}}{T_{SCALED}}\right)^{1.2}$$

Installation weight factors were computed from the equations shown in Table IV. All the installation factors are based on either installed engine airflow or thrust.

TABLE IV

PROPULSION SYSTEM INSTALLATION WEIGHT FACTORS

Air Induction System	W_T	=	.0775	$W_{A_{SLS}}$
Starter	W_T	=	.0075	$F_{N_{SLS}}$
Thrust Reverser:				
Baseline Engine	W_T	=	.0400	$F_{N_{SLS}}$
Adv. Technology Engine	W_T	=	.0350	$F_{N_{SLS}}$
Gearbox, pods, oil tank, etc.	W_T	=	60#	

Where:

W_T = Weight

$W_{A_{SLS}}$ = Airflow (sea level static)

$F_{N_{SLS}}$ = Thrust (sea level static)

Note: Nacelle, pylon and support weights were considered to be a part of the aircraft operating empty weight (owe)

SECTION V

MISSION ANALYSIS RESULTS

1. BASELINE TECHNOLOGY ENGINE/AIRCRAFT SELECTION

In order to conduct a consistent comparative assessment of baseline versus advanced engine technology it was necessary to first define an aircraft configuration incorporating a baseline technology engine. As previously mentioned, the intent was not to conduct a complete parametric optimization for this technology level, but only to define a representative engine/aircraft configuration. The baseline technology engines listed in Table II were investigated for both the primary employment and ferry deployment missions. Table V presents a summary of the aircraft and engine thrust size characteristics for the various base engine thrust/weight ratios considered and Table VI summarizes the resulting aircraft take-off gross weight characteristics.

From Table VI it can be seen that the lowest aircraft take-off gross weights are achieved with engine configurations having the highest uninstalled base thrust/weight ratio considered (7.5:1); and engine overall pressure ratios and bypass ratios in the area of 20:1 and 4.5:1 respectively.

Qualification, within the 1974-75 time period, of a turbofan engine having thrust, cycle and uninstalled thrust/weight characteristics within the range described above is considered feasible based on technology currently being demonstrated by the AFAPL's Advanced Turbine Engine Gas Generator and Advanced Propulsion System Integration development programs. However, the development of such an engine will require that an aggressive development program be undertaken. The characteristics

TABLE V

AIRCRAFT AND ENGINE SIZING CHARACTERISTICS

BPR	OPR	UNINSTALLED SLS/STD DAY * T.C. THRUST	SFC	A/C WEIGHT FRACTION PRIMARY MISSION	A/C WEIGHT FRACTION FERRY MISSION	REQUIRED THRUST SIZE T/W-5.5	REQUIRED THRUST SIZE T/W=6.5	REQUIRED THRUST SIZE T/W=7.5	AIRCRAFT T/W AT T.O.
3.5	20	15372	.447	.7721	.6414	16627	15664	15027	.455
3.5	23	15219	.436	.7776	.6495	16846	15809	15126	.455
4.5	20	13769	.411	.7822	.6563	16573	15618	14999	.457
4.5	23	13617	.398	.7862	.6617	16837	15800	15133	.457
5.5	23	12484	.369	.7900	.6682	16990	15914	15218	.458
6.5	28	11300	.336	.7960	.6788	17675	16376	15560	.457
6.5	20	11678	.355	.7877	.6681	17012	15956	15276	.460
6.5	23	11655	.347	.7921	.6732	17167	16053	15339	.460
7.5	23	10740	.330	.7901	.6871	17570	16349	15574	.460

TIT = 2450°F

* REFERENCE SIZE ENGINE
390 #/SEC SEA LEVEL HOT DAY AIRFLOW

TABLE VI

AIRCRAFT GROSS WEIGHT FOR BASELINE TECHNOLOGY ENGINES

BPR	OPR	AIRCRAFT TAKE-OFF GROSS WEIGHT - LBS BASE T/W=5.5	BASE T/W=6.5	BASE T/W=7.5
3.5	20	147528	138989	133331
3.5	23	149470	140270	134209
4.5	20	146409	137973	132500
4.5	23	148378	139581	132728
5.5	23	149766	140283	134150
6.5	20	149307	140037	134066
6.5	23	150665	140884	134627
6.5	28	156139	144668	137454
7.5	23	154204	143486	136686

of the selected baseline technology engine/aircraft combination are listed below:

SELECTED BASELINE TECHNOLOGY ENGINE/AIRFRAME

Maximum Turbine Inlet Temperature	2450°F
Overall Pressure Ratio	20
Bypass Ratio	4.5
Aircraft Take-Off Gross Weight	132,500 lbs.
Bare Engine T/W	7.5

All advanced technology configurations were compared to the baseline design in terms of relative take-off gross weight (TOGW).

$$\text{Relative TOGW} = \frac{\text{TOGW (Advanced Technology)}}{\text{132,500 lbs (Baseline Technology)}}$$

2. ADVANCED ENGINE TECHNOLOGY

Table VII presents a summary of the aircraft and engine thrust size characteristics for the advanced technology engine configurations investigated during the study. Figure 18 is a carpet plot showing the aircraft gross weight characteristics as a function of various engine parameters. The relative effects of advanced engine technology on aircraft gross weight are shown in Figures 19 and 20. Figure 19 presents relative aircraft take-off gross weight versus bypass ratio, for engine configurations with an overall pressure ratio of 23. This figure also shows the effect of base engine thrust/weight. As shown, engine thrust/weight ratio has a significant effect on aircraft gross weight while variations in bypass ratio have only a minor effect. Section II, paragraph 3, previously indicated the sensitivity of aircraft gross weight to changes in propulsion weight and/or fuel load. The relative flatness of the gross weight versus bypass ratio curve, in general, results from a cancellation of the improved SFC characteristics of the higher bypass ratio cycles by the lower thrust/weight ratio associated with these

TABLE VII

AIRCRAFT AND ENGINE SIZING CHARACTERISTICS

BPR	OPR	UNINSTALLED SLS/STD DAY * T.O. THRUST	SFC	A/C WEIGHT FRACTION PRIMARY MISSION	A/C WEIGHT FRACTION FERRY MISSION	REQUIRED THRUST SIZE T/W=7.0	REQUIRED THRUST SIZE T/W=8.0	REQUIRED THRUST SIZE T/W=9.0	AIRCRAFT T/W AT T.O.
4.5	23	14952	.421	.7840	.6606	14806	14358	14030	.456
4.5	28	14742	.407	.7890	.6663	15100	14576	14211	.457
4.5	34	14414	.394	.7917	.6709	15528	14926	14481	.457
5.5	23	13647	.390	.7900	.6674	14867	14399	14046	.457
5.5	28	13446	.378	.7943	.6738	15109	14588	14211	.457
5.5	34	13118	.368	.7954	.6759	15631	15008	14564	.458
6.5	23	12629	.366	.7940	.6733	14889	14412	14066	.457
6.5	28	12473	.355	.7997	.6825	15079	14555	14177	.457
6.5	34	12283	.343	.8037	.6879	15343	14751	14332	.456
7.5	23	11835	.345	.7976	.6799	15012	14520	14167	.460
7.5	28	11673	.334	.7989	.6848	15266	14718	14342	.459
7.5	34	11435	.323	.8037	.6896	15638	15009	14566	.460
8.5	23	11137	.328	.7973	.6810	14908	14424	14070	.457
8.5	28	10991	.318	.7997	.6861	15280	14732	14342	.460
8.5	34	10729	.309	.7972	.6861	15792	15149	14687	.460

TIT = 2650°F

* REFERENCE SIZE ENGINE
390 #/SEC SEA LEVEL HOT DAY AIRFLOW

TABLE VII (CONTINUE)

BPR	OPR	UNINSTALLED * SLS/STD DAY T.O. THRUST	SFC	A/C WEIGHT FRACTION PRIMARY MISSION	A/C WEIGHT FRACTION FERRY MISSION	REQUIRED THRUST SIZE T/W=8.0	REQUIRED THRUST SIZE T/W=9.0	REQUIRED THRUST SIZE T/W=10.0	AIRCRAFT T/W AT T.O.
4.5	23	15827	.441	.7765	.6509	14659	14301	14030	.458
4.5	28	15649	.428	.7820	.6581	14796	14404	14107	.457
4.5	34	15342	.418	.7845	.6615	15151	14697	14355	.457
5.5	23	14520	.410	.7871	.6629	14433	14089	13828	.456
5.5	28	14390	.396	.7924	.6703	14576	14202	13919	.456
5.5	34	14167	.385	.7964	.6737	14870	14421	14102	.456
7.0	23	12345	.374	.7954	.6761	14452	14097	13810	.457
7.0	28	12840	.362	.8009	.6861	14544	14160	13870	.456
7.0	34	12650	.349	.8048	.6896	14878	14441	14113	.458
8.5	23	11831	.344	.7989	.6819	14485	14129	13860	.459
8.5	28	11727	.333	.8039	.6884	14601	14218	13929	.459
8.5	34	11553	.323	.8069	.6934	14825	14399	14082	.459
10.	23	10925	.322	.7967	.6823	14646	14274	13992	.460
10.	28	10815	.312	.8000	.6879	14754	14353	14050	.459
10.	34	10666	.302	.8047	.6949	14937	14497	14168	.459

TIT = 2850°F

* REFERENCE SIZE ENGINE
390 #/SEC SEA LEVEL HOT DAY AIRFLOW

TABLE VII (CONCLUDED)

BPR	OPR	UNINSTALLED * SLS/STD DAY T.O. THRUST	SFC	A/C WEIGHT FRACTION PRIMARY MISSION	A/C WEIGHT FRACTION FERRY MISSION	REQUIRED THRUST SIZE T/W=9.0	REQUIRED THRUST SIZE T/W=10.0	REQUIRED THRUST SIZE T/W=11.0	AIRCRAFT T/W AT T.O.
7.0	23	13677	.389	.7889	.6666	14136	13867	13656	.456
7.0	28	13560	.377	.7955	.6759	14195	13912	13689	.456
7.0	34	13411	.366	.7995	.6816	14523	14190	13918	.458
8.5	23	12438	.360	.7959	.6768	14083	13815	13589	.457
8.5	28	12364	.348	.8010	.6840	14161	13876	13653	.457
8.5	34	12230	.339	.8047	.6895	14404	14087	13840	.459
10.0	23	11511	.336	.7987	.6821	14196	13922	13706	.460
10.0	28	11440	.326	.8032	.6887	14279	13987	13758	.460
10.0	34	11315	.315	.8084	.6964	14399	14082	13836	.461
11.5	23	10732	.318	.8008	.6873	14323	14035	13809	.460
11.5	28	10680	.308	.8041	.6940	14304	14008	13776	.460
11.5	34	10573	.298	.8067	.6985	14497	14173	13920	.461

TIT = 3050 °F

* REFERENCE SIZE ENGINE
390 #/SEC SEA LEVEL HOT DAY AIRFLOW

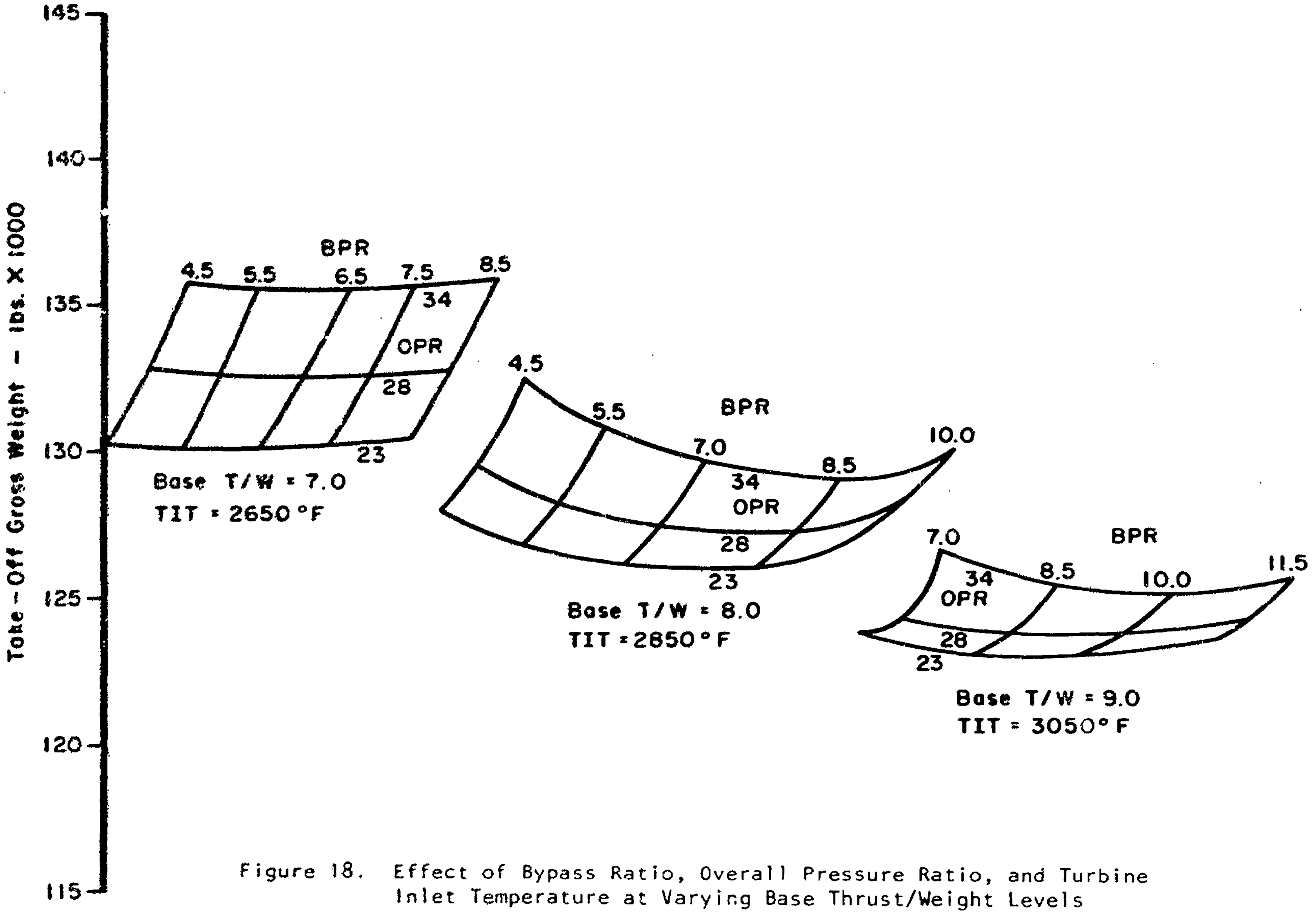


Figure 18. Effect of Bypass Ratio, Overall Pressure Ratio, and Turbine Inlet Temperature at Varying Base Thrust/Weight Levels

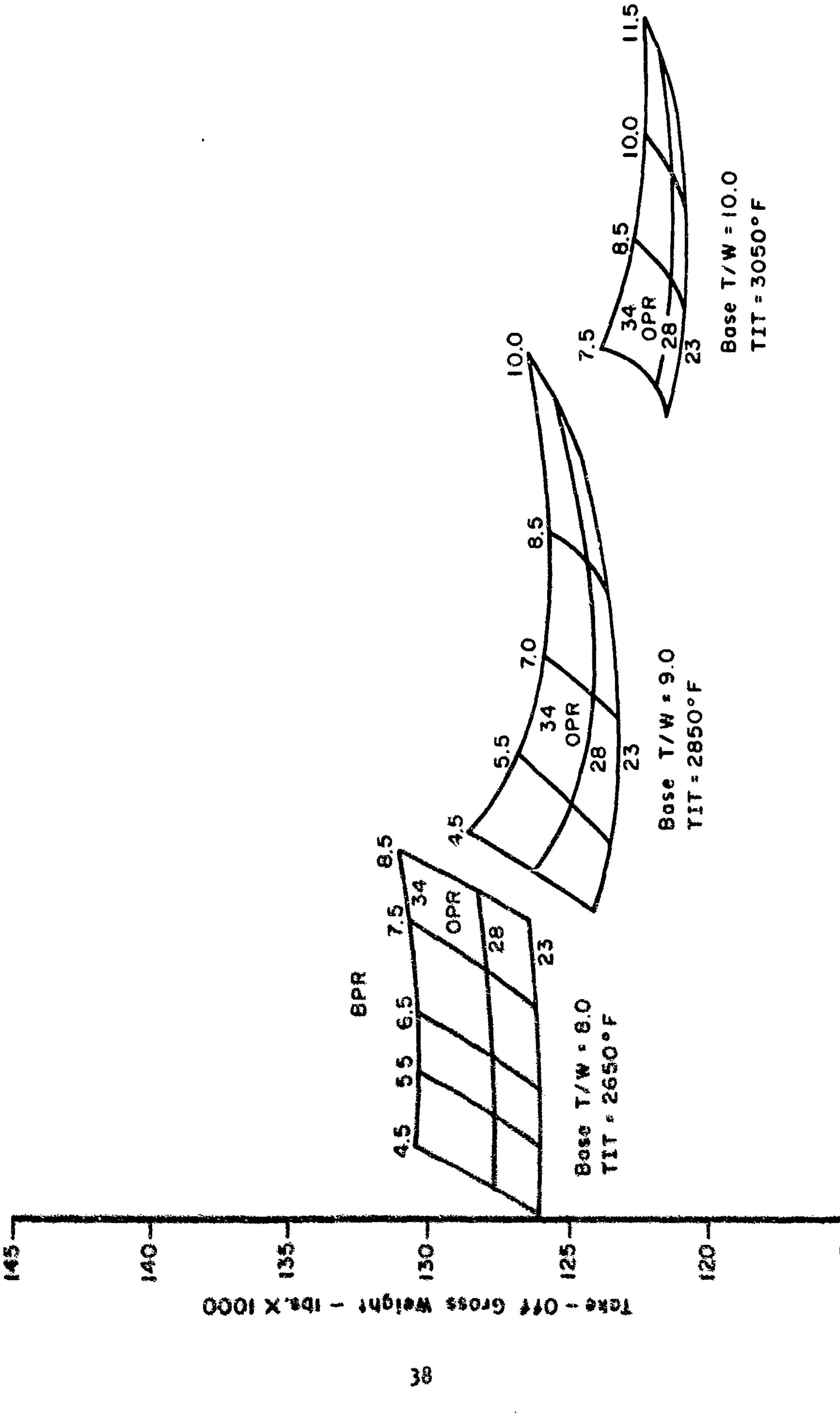


Figure 18 (continued)

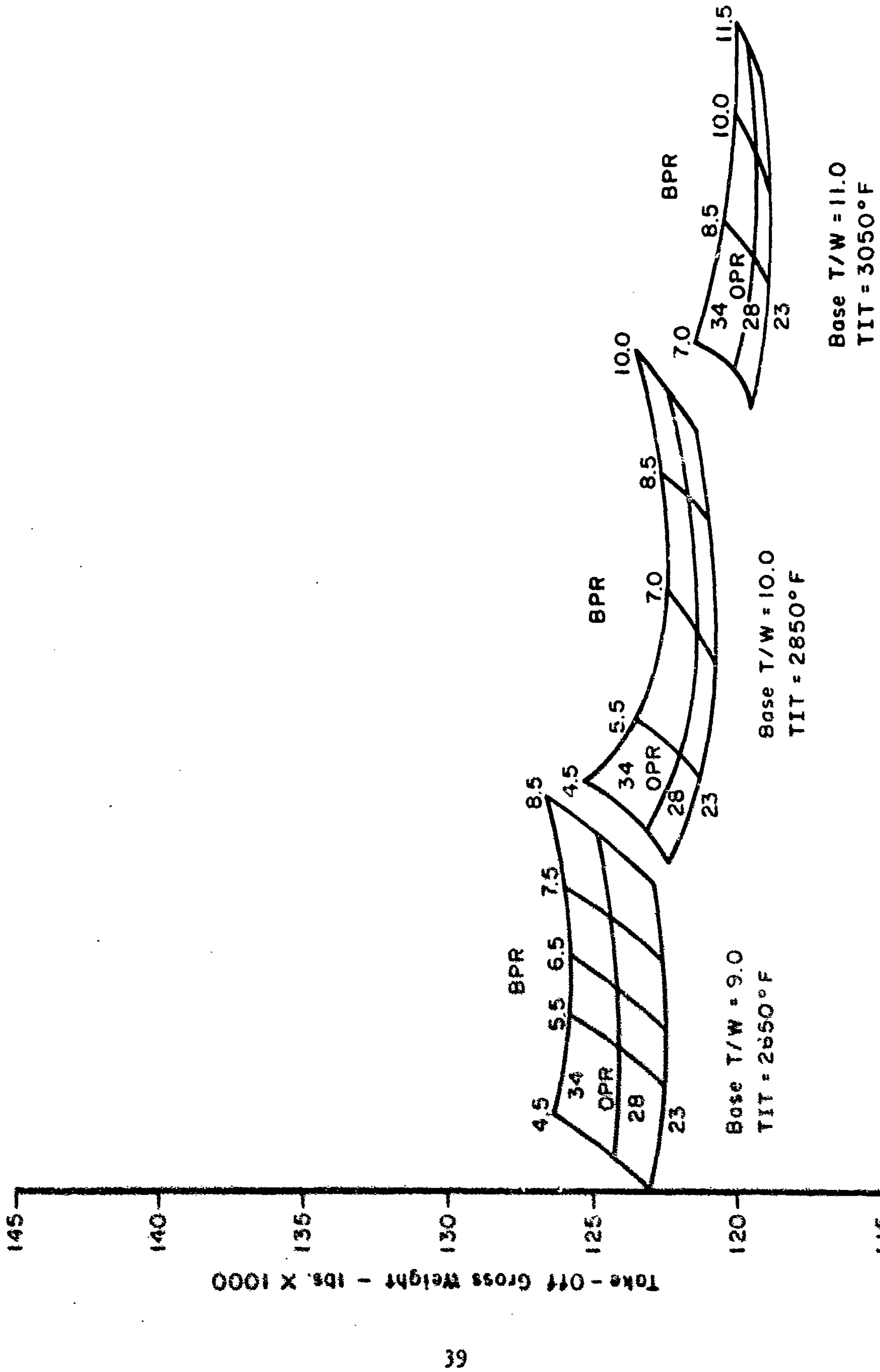


Figure 18 (concluded)

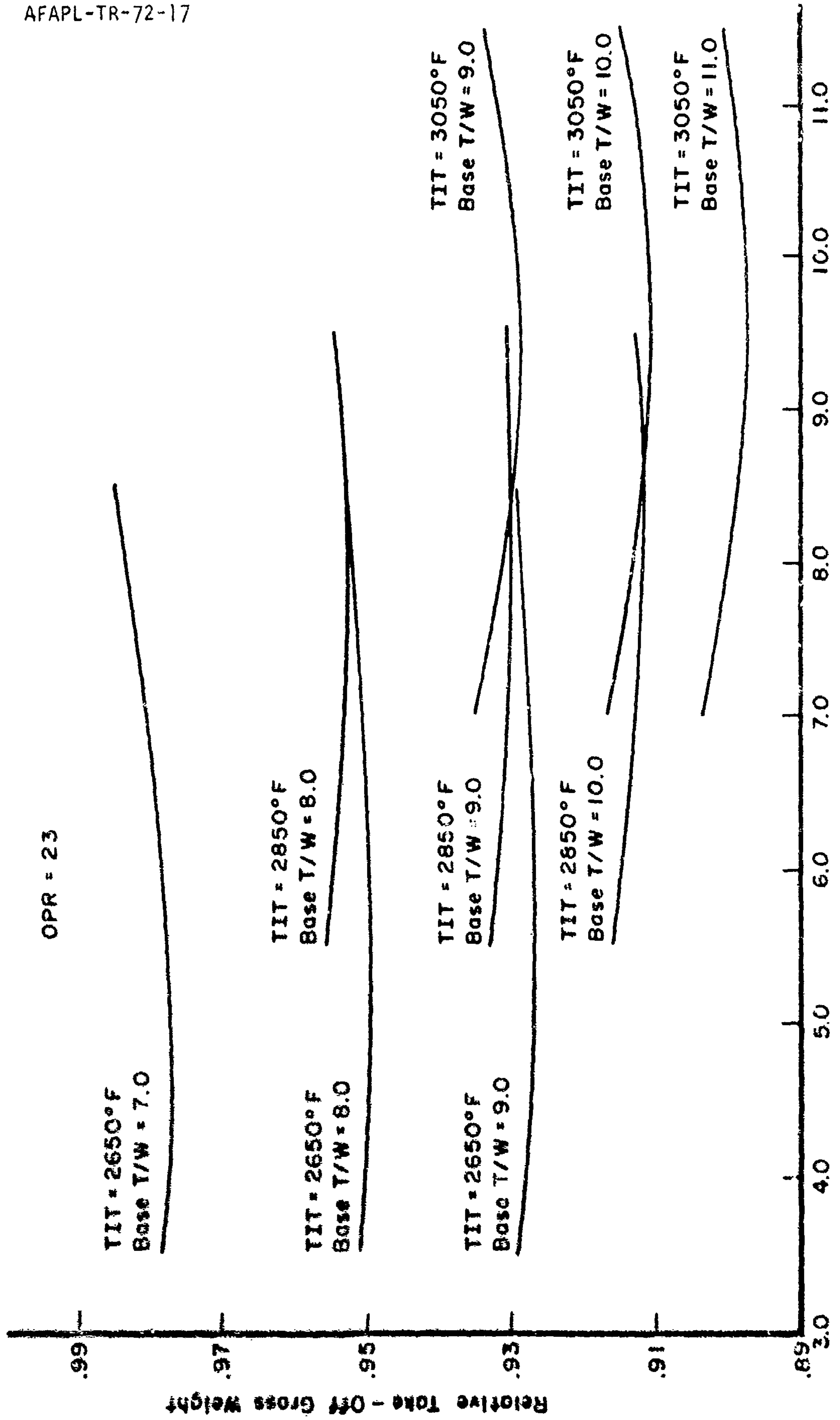


Figure 19. Effect of Bypass Ratio and Thrust/Weight Ratio on Take-Off Gross Weight

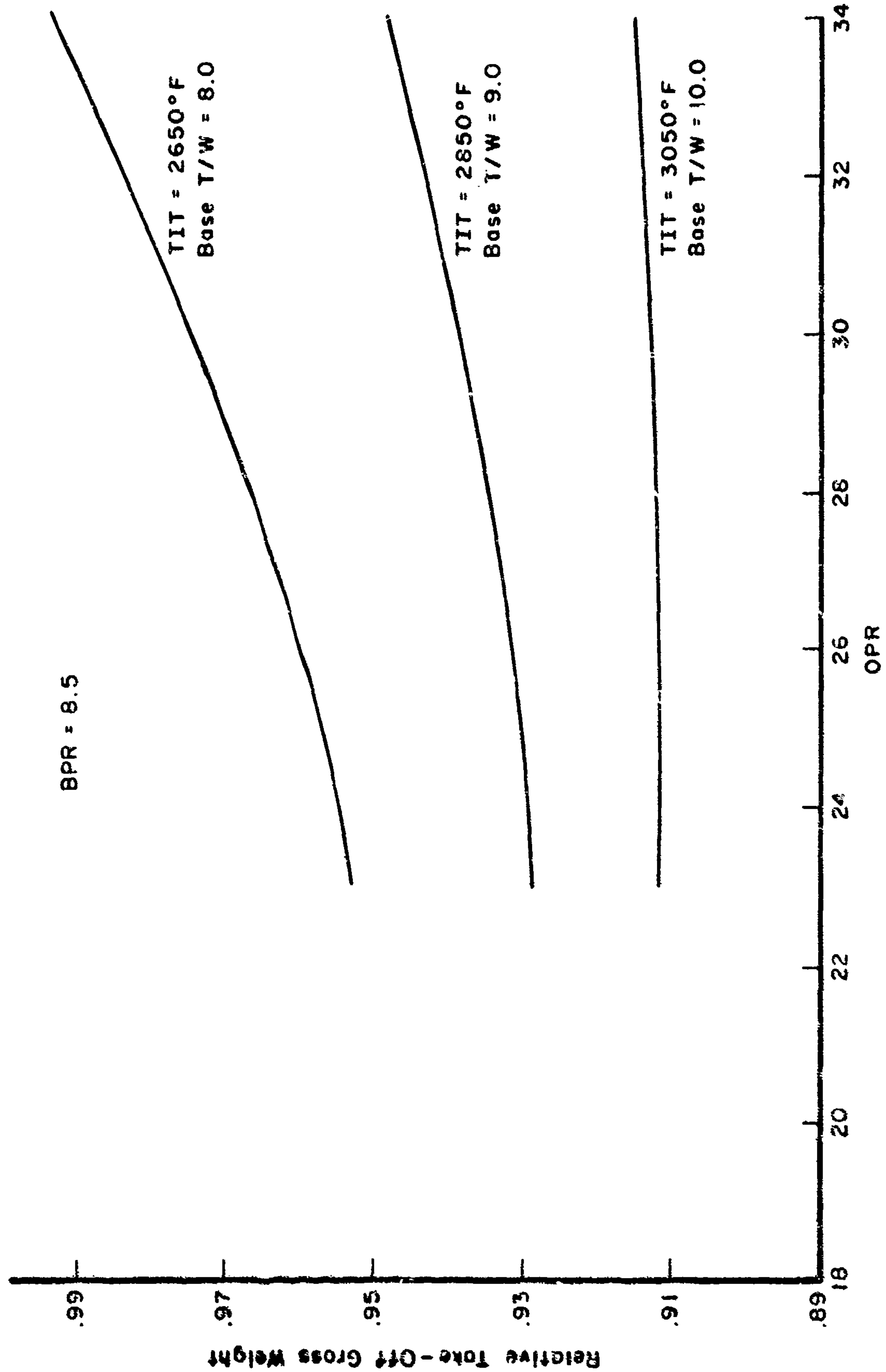


Figure 20. Effect of Overall Pressure Ratio on Take-Off Gross Weight

engines. At a constant bypass ratio of 8.5, Figure 20 presents the effect of varying overall engine pressure ratio, turbine inlet temperature and base engine thrust/weight ratio. Results indicate that moderate overall pressure ratios provide the lowest aircraft gross weight; however, as turbine inlet temperature is increased, for this bypass ratio, the improved performance characteristics of the higher pressure ratio cycles tend to counteract the engine weight increases, resulting in a negligible increase in aircraft weight.

Figure 21 presents the effect of a ±5% change in engine cruise SFC on aircraft weight ~ ~ the primary mission. Figure 22 presents a summary of the relative aircraft take-off gross weight of the optimum engine cycle configurations for each of the turbine inlet temperatures and base engine thrust/weight ratios considered. In essence, this figure can be used to show the effect of engine thrust/weight ratio on aircraft gross weight. Significant relative gross weight reductions are obtained as turbine inlet temperature and base engine thrust/weight ratios are increased for the levels shown in the figure. Figure 23 shows the relationship, for the optimum cycle configurations, of base engine thrust/weight to actual engine uninstalled thrust/weight after corrections for bypass ratio, overall pressure ratio and engine scaling have been taken into account.

3. PAYLOAD SIZING FOR THE FERRY MISSION

Figure 24 provides a comparison of payload weight fraction as a function of turbine inlet temperature and base engine thrust/weight. All the ferry systems depicted meet or slightly exceed the 38,000 pound payload requirement. However, the higher turbine inlet temperature designs with a corresponding increase in engine thrust/weight produce a more effective aircraft in terms of ability to carry payload.

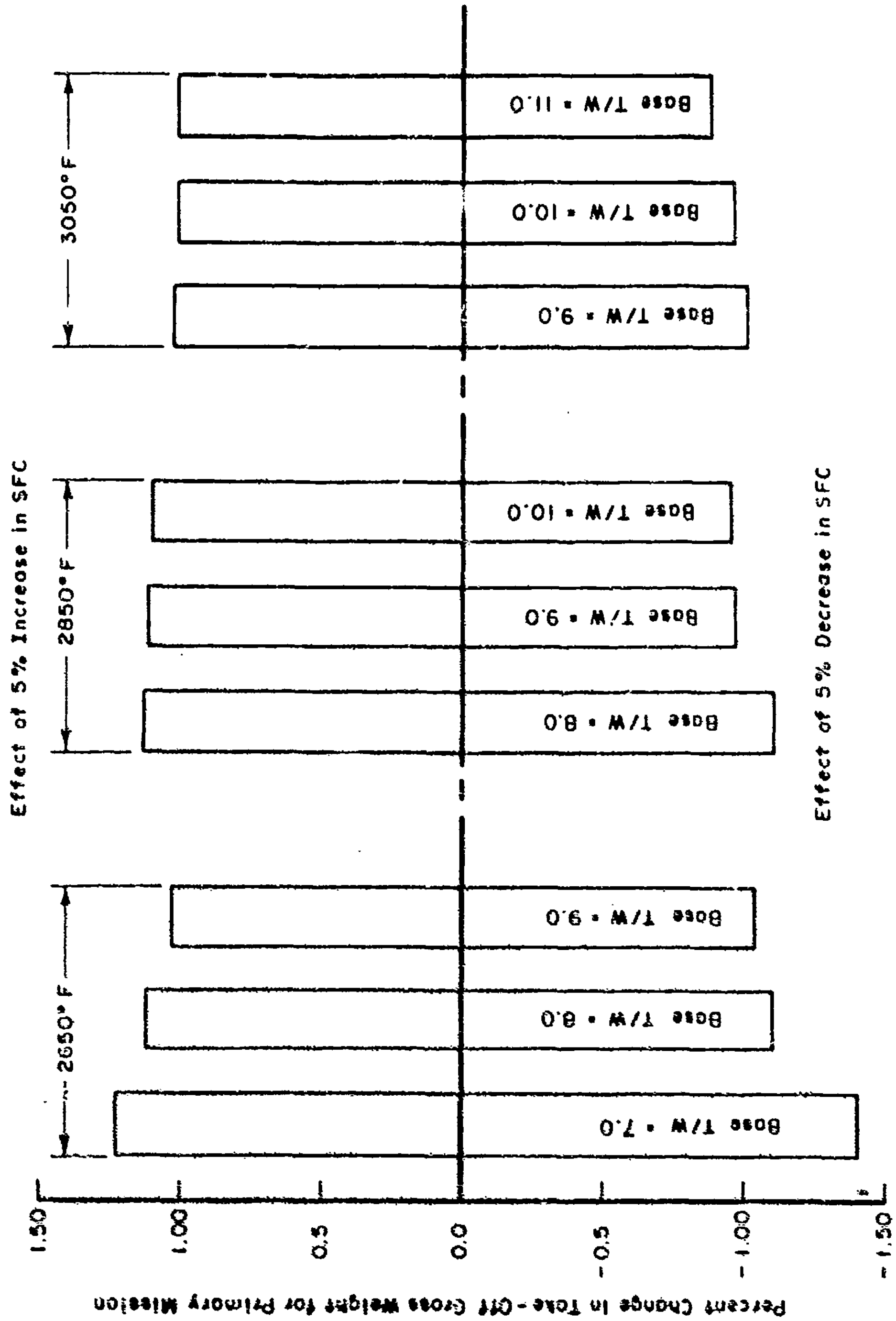


Figure 21. Effect of a 5% Increase or Decrease in Specific Fuel Consumption

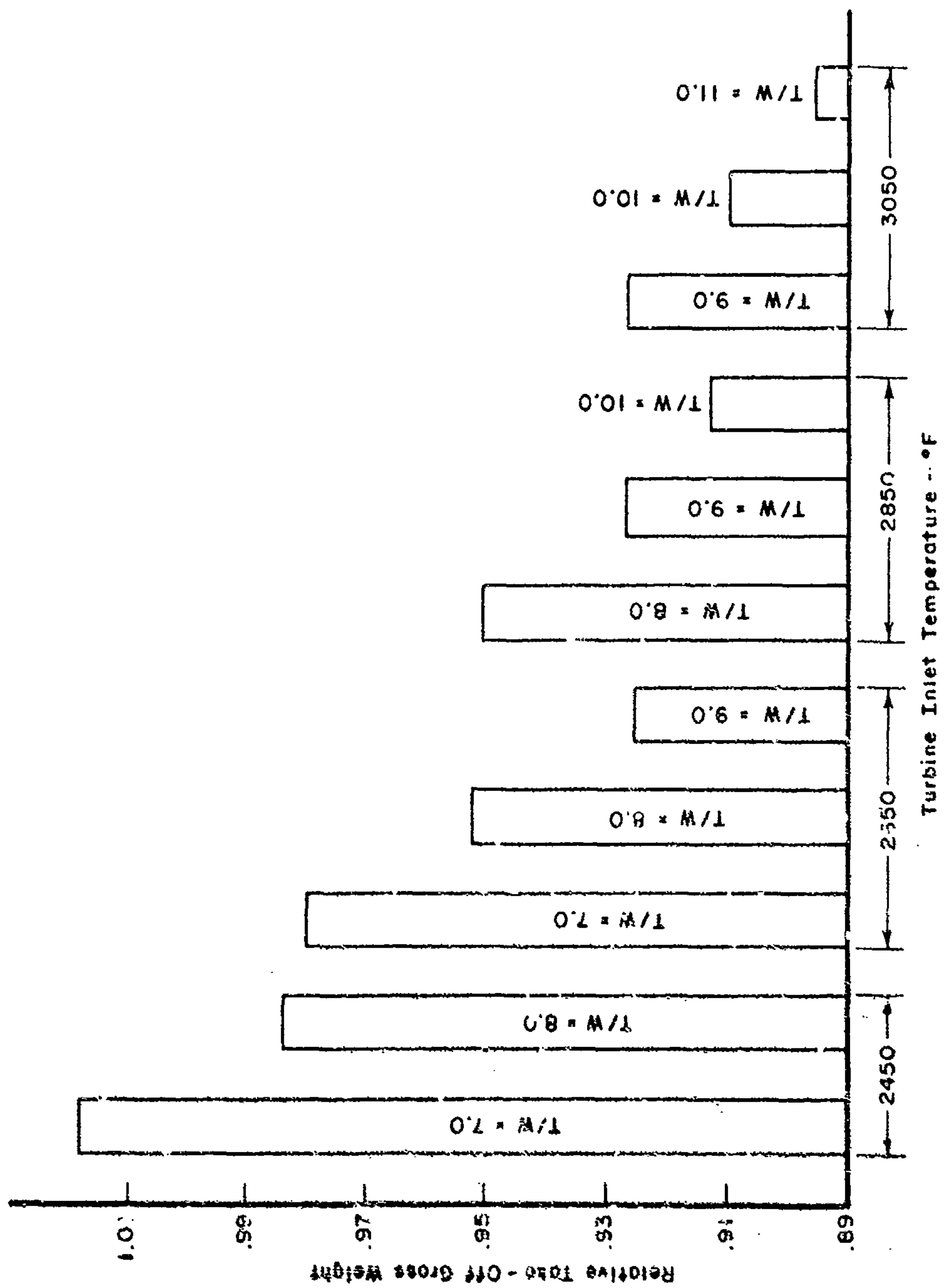


Figure 22. Effect of Turbine Inlet Temperature and Base Thrust/Weight on Aircraft Gross Weight (Optimum Bypass Ratio and Overall Pressure Ratio Cycles)

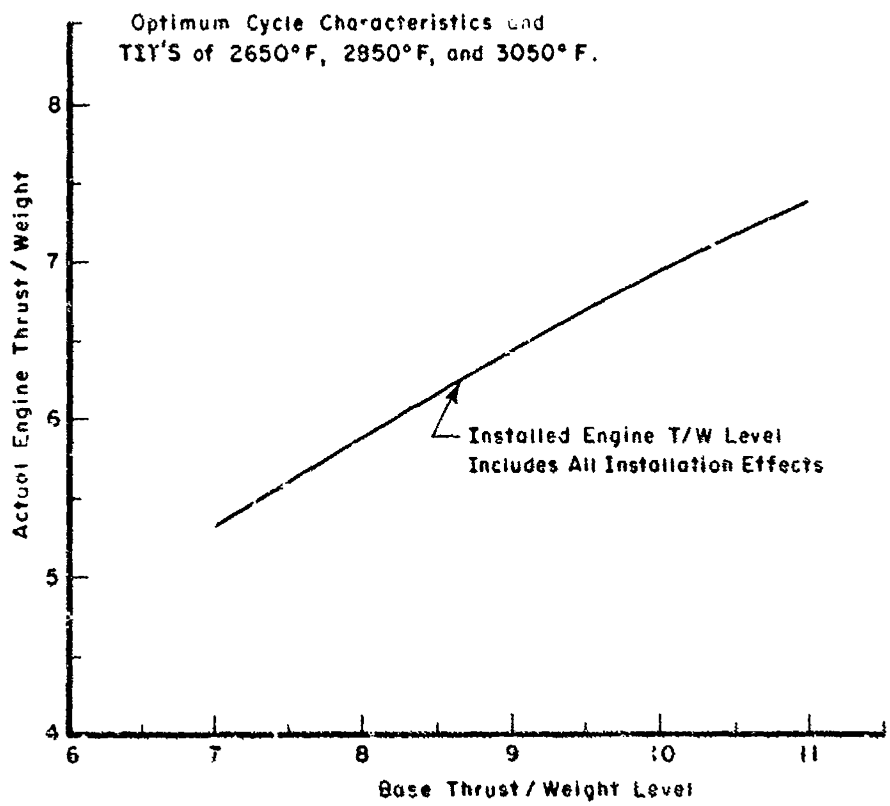


Figure 23. Relationship of Base to Actual Engine Thrust/Weight Ratios

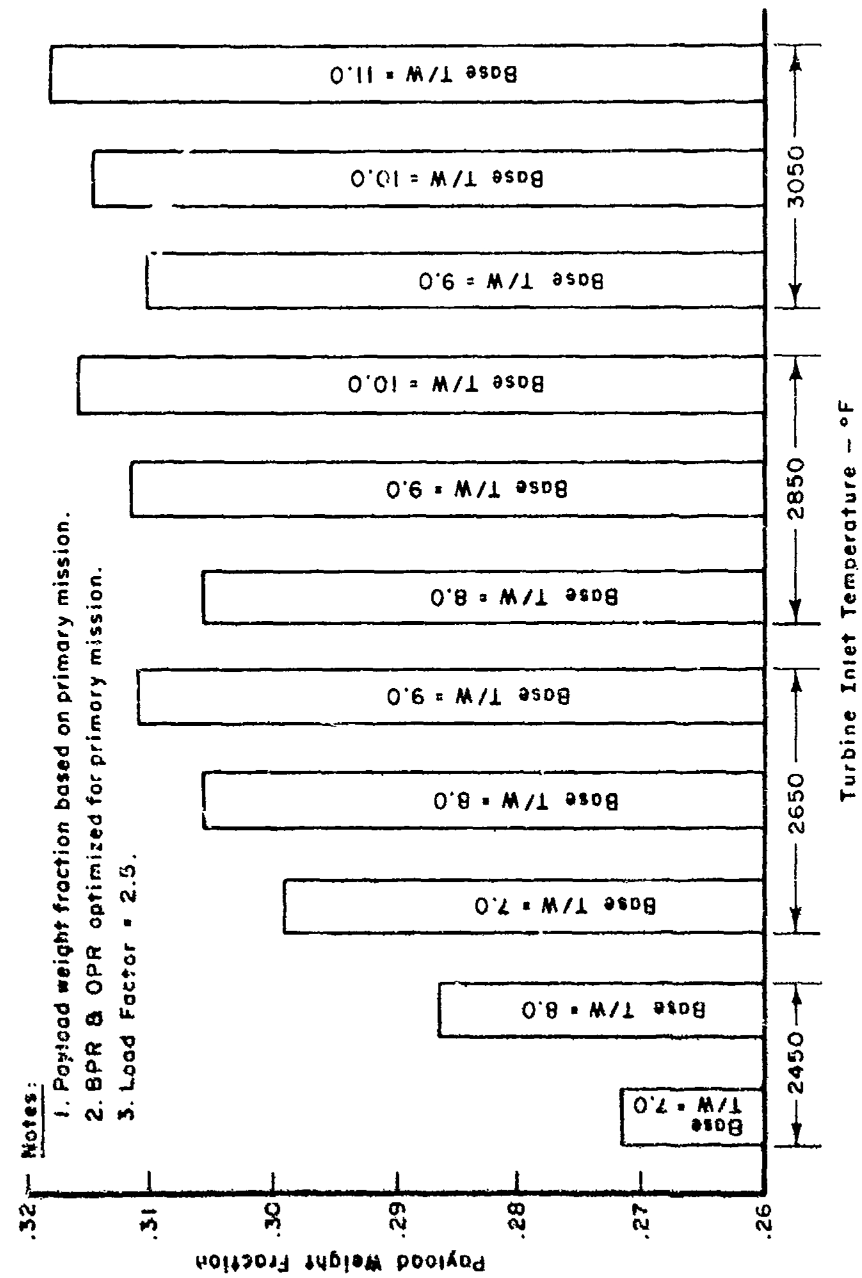


Figure 24. Payload Weight Fraction for Ferry Mission

4. SUMMARY OF RESULTS

Results of this analysis indicate that for the aircraft and mission considered, aircraft gross weight reductions on the order of 10% can be obtained through the application of advanced propulsion technology. Engine thrust/weight, both uninstalled and installed, is clearly the most important engine design parameter.

Engine configurations having moderate overall cycle pressure ratios, from 20 to 28, provided the lightest weight aircraft.

Variations in cruise SFC had only a secondary effect on aircraft weight. Sensitivity investigations of $\pm$ 5% variation in cruise SFC resulted in only a 1 to 1.5% change in take-off gross weight.

Bypass ratios between 3.5 and 11.5, at various turbine inlet temperatures, were considered during the study. Over this range and at any given turbine inlet temperature the maximum aircraft gross weight variation was only two percent. The relative insensitivity of this parameter indicates that if, at a later date, design constraints such as noise abatement or the influence of fan airflow toward improving flap effectiveness becomes increasingly important, engine bypass ratio could be selected so as to favor a particular design condition without significantly affecting aircraft weight.

The ferry mission requirements were met or slightly exceeded by all the engine/airframe configurations based on reducing the allowable load factor from 3.0 to 2.5.

SECTION VI

RECOMMENDATIONS

The study conclusions indicate that, on a parametric basis, significant potential exists for reducing aircraft gross weight through the application of advanced propulsion technology. This technology could be available for aircraft having projected IOC dates in the early 1980 time period. As the first step in ensuring that the required technology becomes available in a timely manner, it is recommended that a propulsion system design analysis be conducted. Having established engine thrust/weight as the most important engine parameter, the principal objective of this analysis should be the definition of turbofan propulsion system installations which offer maximum uninstalled and installed thrust/weight characteristics without requiring significant compromises in overall performance. Definition of engine configurations having moderate overall pressure ratios and uninstalled thrust/weight ratios of at least 9:1 are desired. Strong consideration should be given toward improving the technology associated with the engine low pressure section (fan, fan turbine, fan static structure), since these components currently comprise 55 to 65 percent of the basic weight of high bypass ratio turbofans. In addition, utilization of gas generators having high stage loading rotating components and high maximum turbine inlet temperature capability will minimize the size and weight associated with this section, thereby improving the thrust/weight ratio of the overall engine. During the analysis, the effect of noise abatement on engine performance and weight should be assessed and consideration given to reducing engine noise levels through the use of modified design techniques, engine derating procedures, and/or the utilization of accoustical treatment. Details of the most desirable overall engine and individual component design, performance and weight characteristics should be established during the analysis.

APPENDIX I

GENERAL OUTLINE OF AIRCRAFT AND ENGINE WEIGHT SIZING TECHNIQUE

From the work described in Section IV, both the aircraft take-off thrust/weight and weight fraction are known for each parametric engine configuration. The physical size of the aircraft and the propulsion system are coupled by the expressions on the following page. These equations require that the weight of the aircraft and propulsion system be determined simultaneously; to accomplish this, an iterative procedure was formulated and computerized.

This procedure is based on the arbitrary selection of an engine thrust size and the application of Equations (1) and (2) to determine the accuracy of this selection. The procedure is repeated in a systematic manner until there is satisfactory agreement between the selected thrust and the required aircraft thrust, T_{scaled}. A general outline of the procedure is given below.

1. In order to solve the simultaneous Equations (1) and (2) an installed propulsion system weight must be known.

2. To determine the installed propulsion system weight, an arbitrary thrust size is chosen.

3. The installed thrust/weight for the parametric engine of interest is next determined by means of the following considerations:

a. A basic thrust/weight is selected depending on turbine inlet temperature. Adjustments are then made for OPR and BPR by means of Figures 14 through 17.

b. The thrust/weight is next adjusted for scaling effects which is based on the ratio of the assumed thrust size and the basic thrust size appearing in Table V.

c. Finally, an adjustment is made to account for installation weights, see Table IV.

4. The installed propulsion system weight may be determined from the assumed thrust level and the installed thrust/weight computed above.

5. Equations (1) and (2) are then solved for T_{scaled} using the installed propulsion system weight.

6. If the arbitrary thrust size is not in agreement with T_{scaled}, a new thrust level must be assumed and the process repeated until agreement is reached.

7. When the proper thrust size has been determined, the TOGW can be computed from Equation (1).

$$\mathrm{TOGW}_{\substack{\text{scaled}\\ a/c}} = \Bigg\{ \overbrace{(\mathrm{OWE} + \mathrm{PAYLOAD})\Big|_{\substack{\text{base}\\ a/c}} - 1.87\left(\mathrm{WEIGHT\ PROP\ SYS}\Big|_{\substack{\text{base}\\ a/c}} - \mathrm{WEIGHT\ PROP\ SYS}\Big|_{\substack{\text{parametric}\\ \text{engine}}}\right)}^{\text{ENGINE WEIGHT CORRECTION}}$$

$$- .87\left[\underbrace{\frac{\mathrm{OWE} + \mathrm{PAYLOAD}}{\mathrm{WF}}\,(1-\mathrm{WF})}_{\substack{\text{FUEL ORIGINALLY STORED}\\ \text{IN BASE A/C}}} - \underbrace{\mathrm{TOGW}_{\substack{\text{scaled}\\ a/c}} \times (1-\mathrm{WF})}_{\substack{\text{FUEL NOW ON}\\ \text{BOARD}}}\right]\Bigg\} \frac{1}{\mathrm{WF}}$$

$$T_{scaled} = \frac{\mathrm{TOGW}_{\substack{\text{scaled}\\ a/c}} \times T/W\big|_{a/c}}{\text{NO. OF ENGINES}}$$

WHERE:

OWE - OPERATING EMPTY WEIGHT, BASE AIRCRAFT.
PAYLOAD - PAYLOAD CARRIED THROUGHOUT PRIMARY MISSION.
WF - A/C WEIGHT FRACTION AT THE END OF THE PRIMARY MISSION.
T_{scaled} - THRUST REQUIRED PER ENGINE, AT TAKE-OFF.

The above equations may be simplified to yield:

$$\mathrm{TOGW} = \frac{\mathrm{OWE} + \mathrm{PAYLOAD} - 1.87\,(\Delta\ \mathrm{PROP\ SYS\ WEIGHT}) - .87\,(1-\mathrm{WF}) \times \dfrac{(\mathrm{OWE} + \mathrm{PAYLOAD})}{\mathrm{WF}}}{(1.87\mathrm{WF} - .87)} \qquad (1)$$

$$T_{scaled} = \frac{\mathrm{TOGW}\,(T/W_{a/c})}{4} \qquad (2)$$

APPENDIX II

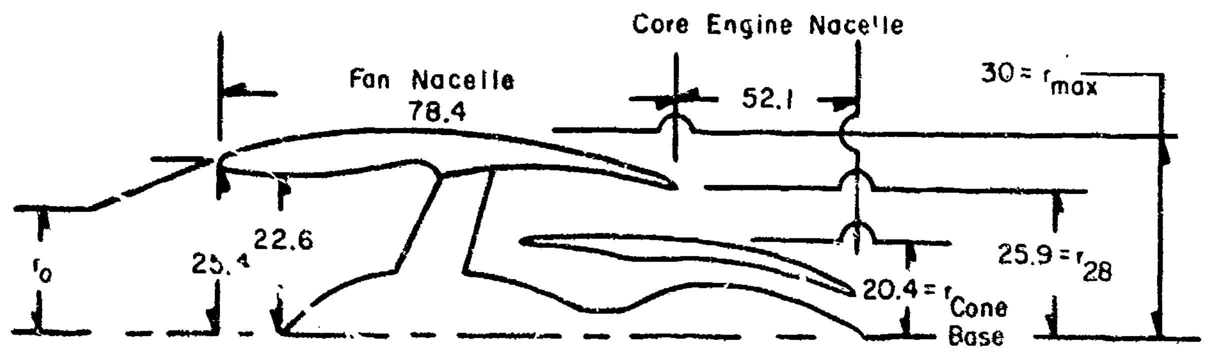
INSTALLED PERFORMANCE PROCEDURES

1. NACELLE SELECTION

A short duct nacelle configuration with an annular conic gas generator nozzle installation was selected because of its low installation losses and lighter weight. See Figure 25.

2. PARAMETRIC ENGINE DIMENSIONS AND SIZING

A reference engine, defined below, whose performance and dimensions were known, was utilized as the base for sizing the parametric engines.



Fan $L/D_{(Ref.\ Eng)}$ = 1.54

Core Engine Nacelle $L/D_{(Ref.\ Eng)}$ = 1.278

Net $Thrust_{(Ref.\ Eng)}$ = 11471 pounds

r = radius in inches

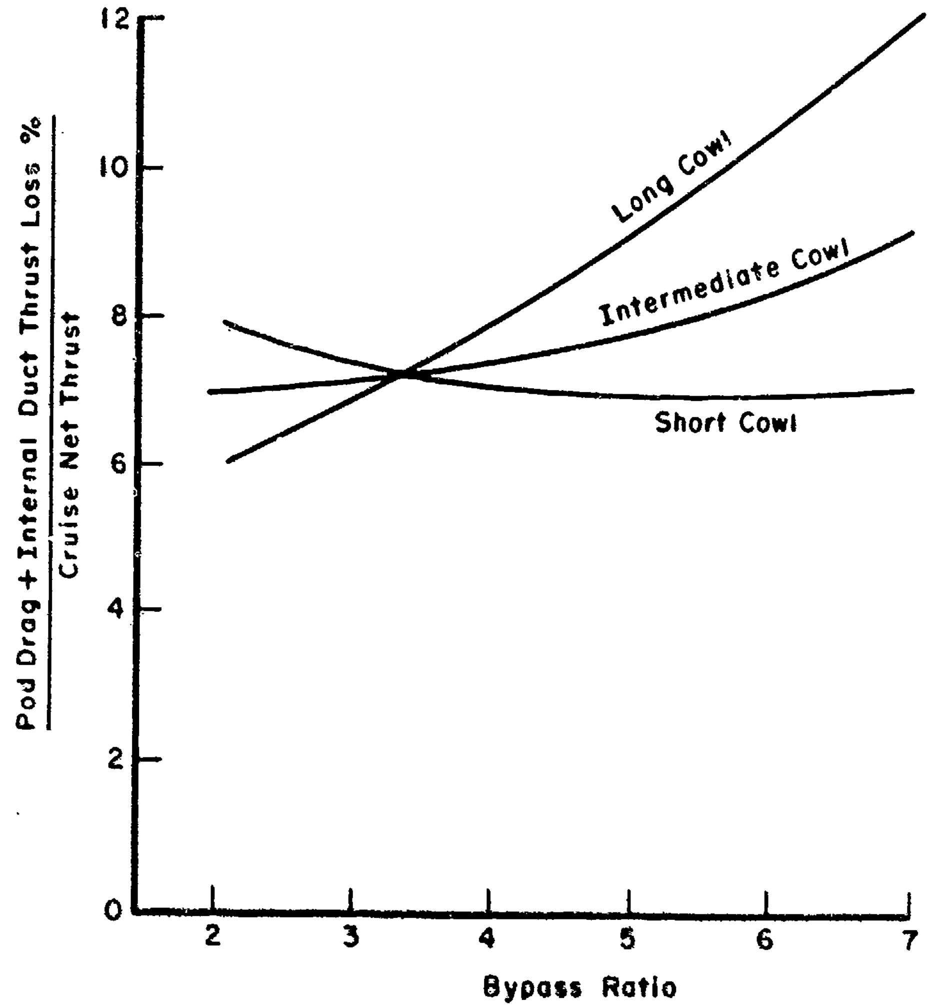


Figure 25. Cowl Length Comparison

The procedure used to size the parametric engine was as follows:

a. The parametric engines were sized to the sea level static thrust of the reference engine.

b. $$A = \frac{\left(\frac{\text{Net Thrust}}{\text{Air Flow}}\right)_{\text{SLS Hot Day T.O. Ref Eng}}}{\left(\frac{\text{Net Thrust}}{\text{Air Flow}}\right)_{\text{SLS Hot Day T.O. Parametric Engine}}}$$

Then: Max Dia Fan Cowl - A (Max Dia of Ref Engine) = B

c. Core Engine Nacelle Gas Generator Base Diameter $$= \frac{\left[(A)\left(\text{Ref Eng Fan Noz Dia}\right)\right]^2 - \left[\text{Fan Nozzle Area Parametric Engine}\right](A^2)}{4} = C$$

d. Length of fan cowl $$= \left[\left(\frac{L}{D_{Max}}\right)_{\text{Ref Eng Fan cowl}}\right] \times \left[D_{Max}\ \text{Fan cowl Parametric Eng}\right] = D$$

e. Length of core eng nacelle $$= \left[\left(\frac{L}{D_{Base}}\right)_{\text{Ref eng core eng nacelle}}\right] \times \left[(\text{Gas Generator Dia})\right] = E$$

f. Area of free stream $$= A_o = \left[\frac{(\text{WA Ref Eng }(R)\ (T_s)}{V_o\ \ (Ps)}\right](A)^2$$

g. Area of gas producer nozzle exit $$\left[\begin{array}{l}\text{parametric engine nozzle}\\ \text{exit area based on ref-}\\ \text{erence engine SLS}\\ \text{Hot Day T.O. } W_A\end{array}\right](A)^2 = A_8$$

h. Area of fan nozzle exit area $$= \left[\begin{array}{l}\text{Parametric engine fan nozzle}\\ \text{exit area based on reference}\\ \text{engine SLS Hot Day T.O. } W_A\end{array}\right](A)^2 = A_{28}$$

i. Area of core engine nacelle $= \pi \frac{(C)}{(2)} \sqrt{C/2^2 + (E)^2}$

j. Area of fan cowl $= \pi \quad (B) \quad (D)$

3. INSTALLED PERFORMANCE

A $C_D \pi$ (Engine Nacelle form plus fan cowl friction drag coefficient) of 0.0185 was used to compute installed performance of all parametric engines. This value appeared reasonable based on available test data. Subsonic inlet additive drag was considered offset by lip suction forces, (Reference 6). Installed performance was computed using a subsonic turbofan installation program, (Reference 4). An additional 0.7% loss was added for pylon drag at the cruise flight condition. Pylon drag at other flight conditions was calculated based on "q" ratio times 0.7%. Interference (engine nacelle/aircraft interface) losses were neglected. Installation losses for the parametric engines are shown in Figure 26.

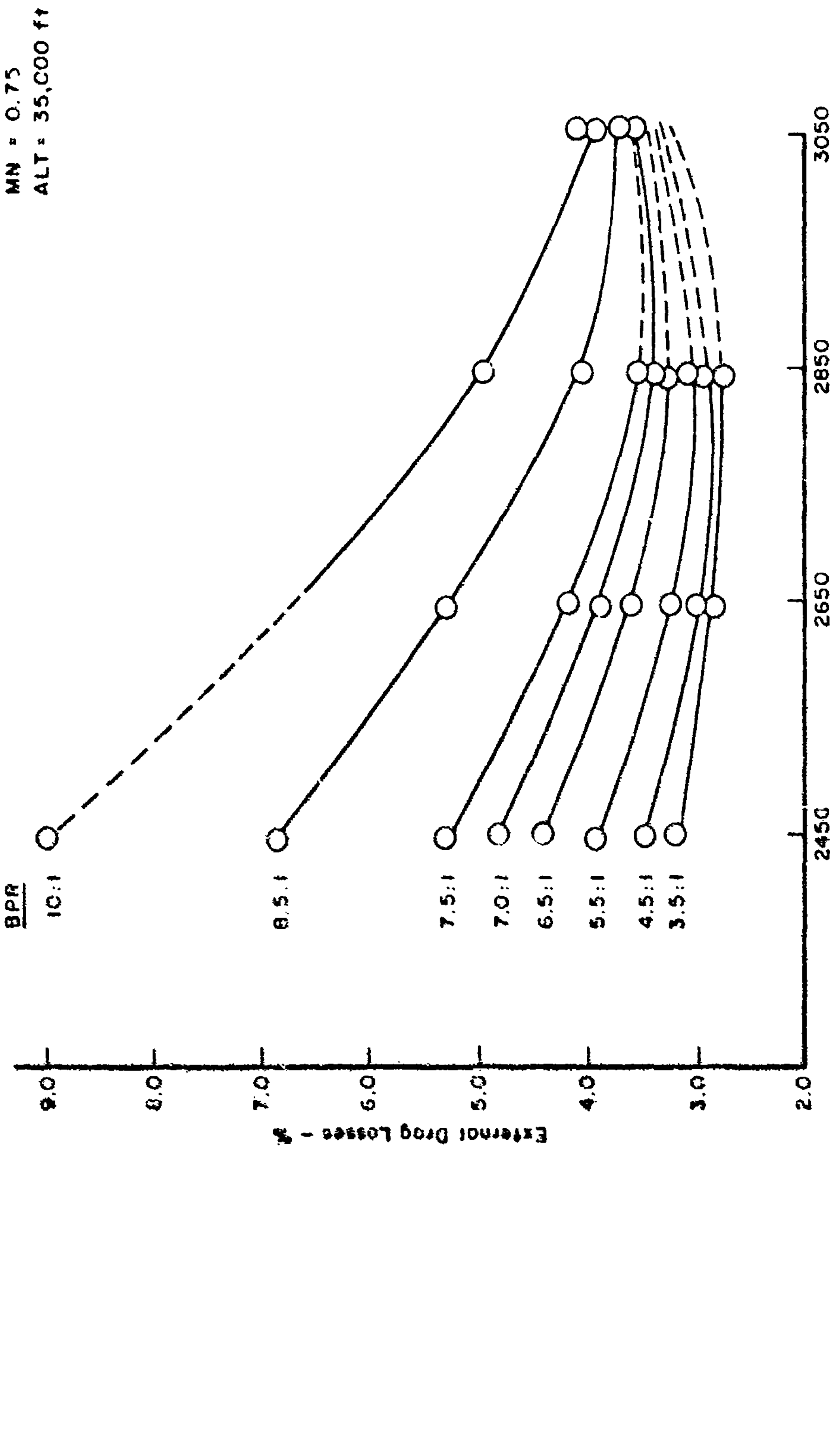


Figure 26. External Drag Losses for Parametric Engines

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UNCLASSIFIED
Security Classification

DOCUMENT CONTROL DATA - R & D	
(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)	
1. ORIGINATING ACTIVITY (Corporate author) Air Force Aero Propulsion Laboratory/TBP Wright-Patterson Air Force Base, Ohio	2a. REPORT SECURITY CLASSIFICATION UNCLASSIFIED 2b. GROUP
3. REPORT TITLE ADVANCED PROPULSION TECHNOLOGY ASSESSMENT FOR AN EXTERNALLY BLOWN FLAP TRANSPORT	
4. DESCRIPTIVE NOTES (Type of report and inclusive dates) Final May 1970 - January 1971	
5. AUTHOR(S) (First name, middle initial, last name) Richard J. Krabal James R. Ruble Raymond N. Leo Carl F. Dienstberger, Jr.	

6. REPORT DATE	7a. TOTAL NO. OF PAGES	7b. NO. OF REFS
May 1972	57	6

8a. CONTRACT OR GRANT NO. b. PROJECT NO. 698DE c. d.	9a. ORIGINATOR'S REPORT NUMBER(S) AFAPL-TR-72-17 9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report)

10. DISTRIBUTION STATEMENT

Distribution limited to United States Government agencies only; test and evaluation; April 1972. Other requests for this document must be referred to Air Force Aero Propulsion Laboratory (AFAPL/TBP), Wright-Patterson Air Force Base, Ohio.

11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY
	Air Force Aero Propulsion Laboratory Wright-Patterson Air Force Base, Ohio 45433

13. ABSTRACT

Results of the study indicate that, for an externally blown flap transport aircraft and missions investigated, aircraft gross weight reductions of 9% to 12.5% can be obtained from the utilization of turbofan engines incorporating Advanced Technology Components. Engine thrust/weight ratio was clearly the most significant propulsion design parameter in terms of providing aircraft weight reductions. Other propulsion parameters such as cruise SFC, bypass ratio, and overall pressure ratio had only secondary effects on aircraft gross weight. While the effect of noise abatement was not considered, variations of engine thrust/weight ratio and cruise SFC were evaluated. Using these variations, preliminary estimates of the penalties associated with noise can be obtained by expressing it in terms of an engine thrust/weight reduction and cruise SFC increase and assessing the resultant aircraft weight increase. A recommendation is made to initiate a preliminary design activity whose objective would be to define suitable, high thrust/weight turbofan propulsion systems for the 1980+ time period.

DD FORM 1 NOV 65 1473

UNCLASSIFIED
Security Classification

14 KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Propulsion Systems						
Turbofan Engines						
Transport A/C						
Advanced Technology Components						